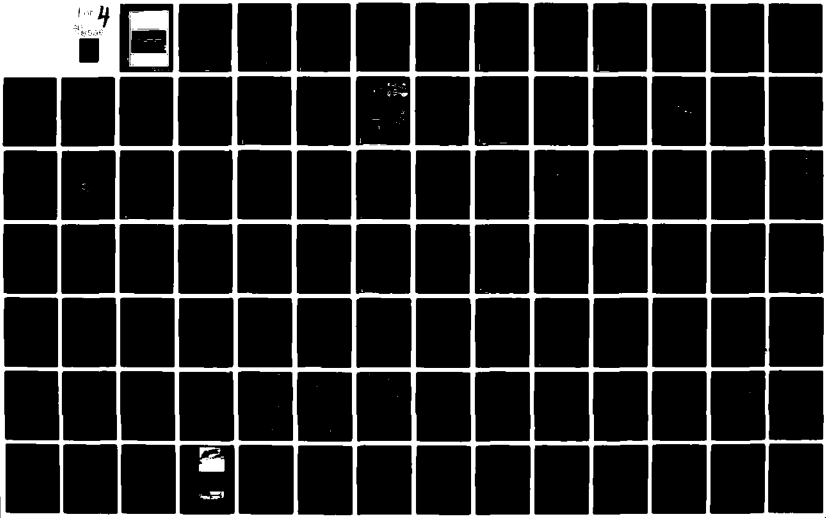
AD-A118 596 ADVISORY GROUP FOR AEROSPACE RESEARCH AND DEVELOPMENT--ETC F/G 1/3
CRITERIA FOR HANDLING QUALITIES OF MILITARY AIRCRAFT.(U)
JUN 82
UNCLASSIFIED AGARD-CP-333 NL



(1)

AGARD-CP-333

AGARD-CP-333

AGARD

ADVISORY GROUP FOR AEROSPACE RESEARCH & DEVELOPMENT

AGARD CONFERENCE PROCEEDINGS No. 333

Criteria for Handling Qualities of Military Aircraft

NORTH ATLANTIC TREATY ORGANIZATION

DISTRIBUTION AND AVAILABILITY ON BACK COVER

NORTH ATLANTIC TREATY ORGANIZATION

ADVISORY GROUP FOR AEROSPACE RESEARCH AND DEVELOPMENT

(ORGANISATION DU TRAITE DE L'ATLANTIQUE NORD)

AGARD Conference Proceedings No.333

CRITERIA FOR HANDLING QUALITIES OF MILITARY AIRCRAFT

Papers presented at the Flight Mechanics Panel Symposium on Criteria for Handling Qualities of Military Aircraft held in Fort Worth, US, 19–22 April 1982.

THE MISSION OF AGARD

The mission of AGARD is to bring together the leading personalities of the NATO nations in the fields of science and technology relating to aerospace for the following purposes:

- Exchanging of scientific and technical information;
- Continuously stimulating advances in the aerospace sciences relevant to strengthening the common defence posture;
- Improving the co-operation among member nations in aerospace research and development;
- Providing scientific and technical advice and assistance to the North Atlantic Military Committee in the field of aerospace research and development;
- Rendering scientific and technical assistance, as requested, to other NATO bodies and to member nations in connection with research and development problems in the aerospace field;
- Providing assistance to member nations for the purpose of increasing their scientific and technical potential;
- Recommending effective ways for the member nations to use their research and development capabilities for the common benefit of the NATO community.

The highest authority within AGARD is the National Delegates Board consisting of officially appointed senior representatives from each member nation. The mission of AGARD is carried out through the Panels which are composed of experts appointed by the National Delegates, the Consultant and Exchange Programme and the Aerospace Applications Studies Programme. The results of AGARD work are reported to the member nations and the NATO Authorities through the AGARD series of publications of which this is one.

Participation in AGARD activities is by invitation only and is normally limited to citizens of the NATO nations.

The content of this publication has been reproduced
directly from material supplied by AGARD or the authors.

Published June 1982

ISBN 92-835-0313-8

Printed by Technical Editing and Reproduction Ltd
Harford House, 7–9 Charlotte St, London, W1P 1HD

PREFACE

A few introductory remarks with respect to the history of the development of handling quality criteria are in order. However, this history will not precede 1971 when an AGARD Specialists' Meeting on the subject was held in Ottawa. At the end of this meeting the Flight Mechanics Panel (FMP) considered appropriate follow-up activities, and one of these was the formation of an ad-hoc committee with an assignment "... to compare the specific Handling Qualities Requirements for aircraft of the various NATO countries and to gather data to validate handling qualities criteria..." Most of the requirements considered by the committee originated in the United States; but all committee members except one were Europeans, and this can probably be explained by the French saying "la critique est aisée mais l'art est difficile" The result of the committee's efforts was AGARD Advisory Report No.89 entitled "Handling Qualities Specification Deficiencies", which was authored by Mr Arthur Barnes from the United Kingdom.

But there were also considerable follow-up activities in the seventies by the workers behind their desks, the flight test engineers and the pilots. In order to disseminate these new results within the NATO countries, FMP devoted a session of its 1978 Stability and Control Symposium, again in Ottawa, to "Criteria for satisfactory behaviour of aircraft with advanced stability and control systems". One of the key questions during the discussions was: Now we have developed sophisticated Control Configured Vehicles with extremely complex stability augmentation and control systems, do we also need new criteria for handling qualities as an aid in aircraft design and certification? Quite a few participants replied, "Yes, of course"; but others were strong advocates of the "Equivalent Systems" methodology, which means in simple (perhaps too simple) words; replace the complex aircraft dynamics by appropriate conventional "make-believe" aircraft dynamics and treat the handling qualities aspects in the classical manner. When the symposium was over, the FMP decided that organizing a full "criteria meeting" would be a desirable follow-up activity and that such a meeting could, at least partially, answer the standard AGARD question as it was posed in 1971 by the Round Table Panel in Ottawa – namely, "Where do we go from here?".

Finally, the ultimate use of advanced aircraft handling quality criteria in the form of specifications for military aircraft must be remembered. Unlike some technical developments that may never be put to practice, the handling quality specifications (e.g. MIL-F-8785C) are indeed used throughout the free world. The superiority of the resulting aircraft depends, in turn, on the quality of these specifications. The theme of the symposium is, therefore, indeed important to AGARD.

The Technical Program Committee for this meeting had a relatively easy task for interest in the subject was considerable, both in Europe and in the United States, and we received many more offers for papers than we could accept. We hope we made the right choices, since decisions were very difficult.

R.O.ANDERSON
J.BUHRMAN
Members, Flight Mechanics Panel

CONTENTS

PRESENT STATUS OF FLYING QUALITIES
CRITERIA FOR CONVENTIONAL AIRCRAFT

by

David J. Moorhouse and Robert J. Woodcock
Flight Dynamics Laboratory
Wright Aeronautical Laboratories
Wright-Patterson AFB, OH 45433

INTRODUCTION

As the first paper at this conference, we decided that a general overview of the subject was required. To that end, the object of the paper is to provide a framework for the remainder of the conference with a general and philosophical discussion of flying qualities criteria relative to the military specification.

Flying qualities research and the development of the flying qualities specification proceeded at a leisurely pace until approximately the mid-1960s. As flight control technology expanded, so did the development of criteria with which to judge the increasingly complex dynamic systems. At the same time the specification lost some of its credibility in the eyes of flight control designers. It appeared frequently that their philosophy was "if an airplane design does not meet the criteria, then the criteria need improving." This attitude goes along with the presumption that the flight control system can cure any problems. As we now know, the new flight control technology also discovered new problems such as phase lag, time delay, etc. We are now at the stage of refining and defining criteria applicable to any future configuration.

In this paper we first trace briefly the development of the U.S. military flying qualities specification up to MIL-F-8785B, issued in 1969. In the late 1960s and 70s many new criteria were proposed, and significant ones are discussed. The equivalent system approach was chosen for MIL-F-8785C and is therefore discussed in some detail. The paper is concluded with our view of future requirements and developments in flying qualities criteria.

BACKGROUND

The Signal Corps Specification[1] for procurement of the Wright Flyer is frequently cited as an ideal. The flying qualities requirement of "perfect control and equilibrium at all times" during flight around a closed course was direct and easy to verify. At that time if the airplane completed the course without crashing the control and equilibrium were more than satisfactory. Difficulty of control was judged by observation and subjectively. By the account of the first military pilot[2] that first military airplane was capable of no task beyond flight itself - which of course was still a great achievement. Reference 2 also documents the first military flight test development program, to modify the Wright Flyer as a stable platform for reconnaissance. In the years that followed, both the aircraft performance envelopes and the piloting tasks expanded rapidly. In this country, our first record indicates the start of codification of flying qualities in 1940 by Hartley A. Soule at NACA,[3] based on Edward Warner's DC-4 experience. Flying qualities research was conducted mostly by NACA until a substantial data base of acceptable and unacceptable flight characteristics was available by the end of WWII. In 1943 the U.S. Army Air Forces issued their first "flying qualities" specification.[4] This specification listed acceptable stability and control characteristics in prescribed flight test maneuvers. Coordination was soon achieved with the U.S. Navy, and this same format was maintained through a few revisions. William H. Phillips' classic NACA report[5] should be mentioned. MIL-F-8785B,[6] issued in 1969, represented a significant change. The response requirements were expressed in terms of named classical modal parameters, natural frequency, damping ratio and time constant. In this form it was most responsive to the design process, but unfortunately it conveyed the unintended impression that it applied only to the dominant roots of the airplane dynamics. As indicated in Reference 2, achievement of good mission-oriented handling qualities has been a problem since the beginning of flight. And for just as long (viz. the Wright's wing warping-to-rudder interconnect prior to 1903) designers have sought solutions by various means, more or less successful, involving the flight control system. In his comprehensive 1949 textbook on stability and control[7] Courtland Perkins discusses aerodynamic balance of control surfaces, a variety of tabs to improve stick-force stability or reduce hinge moments, bobweights, downsprings and other "gadgetry." As hinge moments became less and less tractable, first hydraulic boost (F-94/T-33 ailerons) and then fully powered controls with no effective mechanical reversion (F-89) came into use. This "progress" in flight control technology was needed in order to utilize the extended performance envelopes of new aircraft, but created problems of its own. See, for example, the discussion of bobweights and pilot-induced oscillations in Reference 8. With fully powered controls came the possibility of improving the aircraft characteristics apparent to the pilot through series stability augmentation, which does not move the pilot's cockpit control; early examples are the F-89 and B-47 yaw dampers. The years since have seen the use of flight control technology in more and more ways so that actual airplane dynamics became more complex. Thus grew the perception that the specification did not apply. The so-called flight control technology explosion fostered a criteria explosion: stability and control augmentation systems (SCAS) with "response control," forward and feedback-loop compensation, prefilters, digital mechanization, etc.

Reference 9 is a landmark guide to the 1969 flying qualities requirements, prepared at the insistence of the Air Force Flight Dynamics Laboratory's C. B. Westbrook. Although there was some reluctance to show how meager was the basis of some requirements, these 689 pages have proven to be an invaluable compendium of the rationale and data base for specification.

SUMMARY OF CRITERIA

First we will discuss the flying qualities criteria which have been proposed either as alternatives to the specification or as guides for a particular design. As will be seen, the majority are for the short-term pitch tracking task.

Numerator Time Constant

The well-known approximation for the classical short-term response of pitch rate to control inputs is

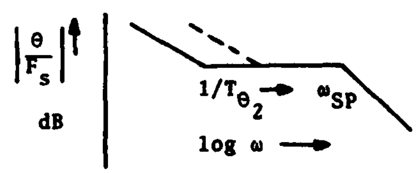
$$\frac{q(s)}{F_s(s)} = \frac{M(s + 1/T_{\theta_2})}{(s^2 + 2\zeta\omega_n s + \omega_n^2)}$$

All along, there has been considerable discussion as to whether T_{θ_2} or n_α is the more appropriate parameter to characterize pitch response (n_α being the steady-state ratio of normal acceleration response to angle-of-attack response for pitch control inputs). References 9 and 10, for example, both discuss the importance of the numerator time constant, while Reference 11 suggests that it is indeed the governing parameter in landing approach. The British flying qualities specification (Reference 12) uses the ω_n^2/n_α requirements of MIL-F-8785B/C, except that the Category C (terminal flight phases) requirements are modified by the addition of the following:

Minimum values of n_α	$V < 100$ kn	$V > 100$ kn
Level 1 boundary	1.67	V (kn)/60
Levels 2 and 3 boundary	1.0	V (kn)/100

With the classical approximate relationship $n_\alpha = (V/g)/T_{\theta_2}$, this is equivalent to specifying maximum values for T_{θ_2} of 3.1 secs for Level 1 and 5.2 secs for Levels 2 and 3. There is also a note that these lower bounds of n_α may apply to Category A (the more demanding of the piloting tasks in up-and-away flight) as well as Category C.

The requirements of MIL-F-8785B were in terms of n_α and did not consider numerator time constant per se. It is possible for two classical aircraft to have the same values of short-period frequency and damping and n_α but different numerator time constants, just by virtue of having different airspeeds. The specification would not discriminate between these two configurations in terms of flying qualities, although there is a change in pitch dynamics, as in the asymptotic frequency response sketched. The effect of that change on pilot opinion rating is going to be influenced by the proximity of $1/T_{\theta_2}$ to the short-period frequency (and possibly by the phugoid dynamics also). To support using n_α instead of $1/T_{\theta_2}$ for the final version of MIL-F-8785B, Reference 8 cited:



- $\omega_{n_{SP}}^2/(n/\alpha)$ and ζ_{SP} boundaries fit the available flight data, over an n/α range of 12.3 to 61.5 for Category A Flight Phases, roughly 2 to 20 for B-70 Category B, and 2+ to 11 Category C.
- $\omega_{n_{SP}}^2/(n/\alpha)$ corresponds to $(F_s/n)M_{F_s}$ and to Bihrle's Control Anticipation Parameter, Reference 12, the ratio of inital pitching acceleration to steady-state load factor, for pitch control. This correspondence holds for most any form of stability augmentation involving only the pitch control, as well as for the basic airframe.

Also $\omega_{n_{SP}}^2/(n/\alpha)$ tends to be invariant with speed, so that over a wide speed range an airplane can stay within the boundaries. That is a nice practical convenience.

Recall that Bihrle's Control Anticipation Parameter, CAP, (Reference 13) is :

$$CAP = \frac{\dot{q}_o}{n_\infty} = \frac{s^2 N_\theta|_{s\to\infty}}{N_n|_{s\to o}} \approx \frac{\omega_{SP}^2}{n/\alpha} \approx \frac{g}{V}\,\omega_{SP}^2\, T_\theta$$

The equivalence of CAP and ω_{SP}^2/n_α holds fairly generally unless an additional control

surface (such as DLC) is available to alter the characteristics. An interpretation of CAP is that the ratio of initial pitch acceleration to steady state normal acceleration defines the compatibility of the flight path response and the initial sensation of a pitch control input. An alternative CAP[14] uses initial cockpit normal acceleration instead of $\dot{q}_o$ since at the pilot-location:

$$n_{po} = n_o + x_p \dot{q}_o$$

This interpretation seems to fit some large aircraft, for which pilot location can be expected to have a significant effect. There is then the problem of explaining past successful combat aircraft with zero or negative x_p. Modifications to account for higher-order effects have been proposed by Chalk[15], and by Bischoff[16] who achieves good correlation with a number of Navy aircraft.

From the asymptotic frequency-response sketch of $|\theta/F_s|$ it is the frequency separation of $\omega_E T_E$, that determines the degree of departure from "ideal" k/s-like response at frequencies below ω_{sp}, and thus possibly the gain margin:

$$\log \omega_E - \log 1/T_E = \log (\omega_E T_E)$$

so that from consideration of the pitch response alone, the proper parameter would be $\omega_E T_E$, as proposed for possible revision of MIL-F-8785C (Reference 17). Nevertheless, Reference 8 shows that the similar parameter $\omega_E^2 T_E$ g/V correlates the available data; Reference 15 shows further correlation.

In summary, all the preceding variations are attempting to specify the essential charactertistics of a 1st/2nd order response. As the discussion implies, for aircraft which do exhibit this form of response the actual combinations of parameters do not matter too much. Problems that have arisen have been in analysing responses which are not truly of this form.

Bandwidth and Phase Sensitivity

For Category C Flight Phases Ashkenas, Hoh and Craig (Reference 18) propose Level 1 and 3 requirements (Fig. 1) to assure adequate pitch attitude bandwidth. In this early application is already recognized the need for a measure of phase sensitivity, the variation of phase as frequency increases from the bandwidth frequency.

This requirement gets directly at "The basic inner attitude response features which are necessary regardless of outer-loop control problems or auxiliary (e.g., direct lift) control." It applies to "The complete airplane attitude response including both the phugoid and short-period modes, ... flight control system characteristics (and) the various controlled element forms resulting from current flight control augmentation concepts." However, we saw sufficient drawbacks not to use it. There is no Level 2 boundary, and the data points shown with pilot rating $\leq$ 6.5 were scattered on both sides of the "Level 3" boundary. In addition, recent experience (e.g., Ref. 19) indicates that a 1 rad/sec.bandwidth is often insufficient for the flare and touchdown phase of a precision landing. Also there is a natural reluctance to have such different forms of requirement for terminal and up-and-away flight.

Ralph Smith's Criteria

Ralph Smith (Ref 20) proposes a set of requirements for short-term longitudinal response based on a "no-tracking hypothesis:" "Optimum handling qualities demands minimum closed-loop control by the pilot." His parameters include:

- t_q — time to first peak of the q(t) response to a step input in stick force
- $\phi(j\omega_c)$ — $\measuredangle \frac{a_{z_p}}{F_s}(j\omega_c) - 14.3\,\omega_c$; a_{z_p} is normal acceleration at pilot station
- ω_c — criterion frequency, rad/sec, approximately the crossover frequency of the pilot - aircraft system dynamics for pitch attitude tracking; a function of aircraft dynamics and disturbance bandwidth (Fig. 2)

His proposed requirements are:

$$0.2 \leq t_q \leq \text{ for Level 1}$$

$$S \triangleq \frac{d}{d\omega}\left|\frac{\theta}{F_s}(j\omega_c)\right| < -2\text{db/octave for Level 1}$$

$$\measuredangle \frac{\theta}{F_s}(j\omega_c) \geq -123^\circ \text{ for Level 1, } -165^\circ \text{ for Level 2}$$

Smith further states that when controlling pitch attitude, if closed-loop damping is insufficient, a pilot may switch to normal-acceleration control. In that case phase

margin of $n_z(j\omega)/F_s(j\omega)$, evaluated at the θ/F_s crossover frequency, is an indicator of pilot-induced oscillation (PIO) tendency. His PIO criteria are:

$$\phi(j\omega_c) \geq -160^\circ \text{ when } -122^\circ \geq \measuredangle \frac{\theta}{F_s}(j\omega_c) \geq -130^\circ, \text{ for Level 1}$$

$$\phi(j\omega_c) \geq -220^\circ \text{ when } -148^\circ \geq \measuredangle \frac{\theta}{F_s}(j\omega_c) \geq -165^\circ, \text{ for Level 2}$$

Level 3 floors exist, but data to establish them are lacking. This set of requirements was proposed tentatively, subject to further validation. Smith proposes similar requirements for direct-lift control modes and for tasks in which relative position is important, such as aerial refueling and formation flight. Time did not permit full consideration of Smith's suggestions for MIL-F-8785C.

Time-Domain Criteria

The time history of a given second-order transfer function is completely specified. Again, as aircraft became more complex many researchers felt that the time history contained the parameters influencing the pilot. Accordingly several time-domain criteria have been proposed.

C*: With the thought that pilots are relatively more interested in pitch rate at low speed but normal acceleration at high speed, Malcolm and Tobie (Ref. 21) proposed a criterion in terms of the parameter

$$C^* = K\left(n_z + \frac{V_{co}}{g} q + \frac{l_p}{g}\dot{q}\right)$$

where $n_z + l_p\dot{q}$ is the normal load factor at the pilot station and V_{co}, often taken to be 400 ft/sec., is the airspeed at which the n_z and q signals are equal. Malcolm and Tobie derived C* time-history boundaries from Cornell Aero Lab $\omega_n^2 - 2\zeta\omega_n$ "bull's-eyes"(see Ref. 8, p. 63). Later Kisslinger and Wendl proposed modified C* boundaries (Ref. 22) derived from their ground-based simulator studies. They also extended the concept to propose analogous parameters for the lateral and directional axes: D*. Time-history bounds are an appealing form of criteria, useful to the flight control designer. However, several investigators (e.g., Neal and Smith[23] and Brulle in a McDonnell internal memo dated 31 December 1974) have found the C* criterion lacking in good correlation with pilot rating of flying qualities.

While pilots do not characteristically make the step control inputs used in this and a number of time-response criteria, a step does have a broadband frequency content, though amplitude varies with frequency. Malcolm and Tobie also devised a frequency-response version of their C* criterion.

Time Response Parameter (TRP): Abrams' TRP (Ref 24) is based on dead time, τ ; delay time, t_d; cyclic time, t_c; and ratio of overshoot to steady state, A_1, for the pitch-rate and normal-acceleration responses to a step stick force:

$$TRP = (TRP)_{\dot{\theta}} + (TRP)n_z + 0.2\ (\tau_{nz} - 0.2)$$

$$(TRP)_{\dot{\theta}} = (t_d/t_c)_{\dot{\theta}} + 0.08\ (A_{1_{\dot{\theta}}} - 1.0)$$

$$(TRP)_{nz} = 0.5\ (t_{d_{nz}} - 0.7) + 0.3\ (A_{1_{nz}} - 0.3)$$

where the constants were determined empirically. The $0.2\ (\tau_{nz} - 0.2)$ term is used only when TRP is small, less than 0.23. All terms must be positive; if any should be negative they are set to zero.

Brulle and Moran (Ref. 25) plot this criterion using the data of Ref. 26 to show good correlation with Cooper-Harper rating:

$$PR = 10 - 12.19 \exp(-3.18\ TRP)$$

with ± 1 rating encompassing almost all the data. Using fixed-based simulator evaluations, Brulle again gets excellent correlation of TRP with pilot rating. However, Figure 3 shows this trend to be rather different from that of the Di Franco data. Some moving-base simulator results were intermediate, as were some cases which had deadbeat response with and without direct lift. For these Abrams has suggested a modified TRP with an additional term

$$TRP_{DB} = 1.4\tau_{nz} + .16$$

Thus TRP appears to be a useful indicator of flying quality trends, though it does not yet seem definitive enough to use as a requirement.

Chalk's Pitch Rate Response Criteria: In Reference 27, Chalk proposed requirements on pitch rate response as shown in Figure 4. Maximum values for effective time delay, t_1, were also specified but since they are similar to the requirements in 3.5.3 of MIL-F-8785C they are not discussed here. For a classical second-order system the parameters used, transient peak ratio and rise time parameter, are directly related to the parameters used in MIL-F-8785C[28], viz. damping ratio and Control Anticipation Parameter. Once formulated as shown, however, the requirements are independent of systems order and apply directly to the actual response - thus avoiding problems of interpretation. The actual numbers themselves are also revised from the corresponding ones in MIL-F-8785B: lower Level 2 and 3 boundaries for damping ratio and lower boundaries at all levels for ω_n^2/n_α (Chalk places no Level 3 requirement on the rise-time parameter).

This is one of the requirements considered in our recent study[29] of flying qualities for large airplanes.

Neal-Smith Criterion

A criterion for good closed-loop pitch tracking was proposed by Neal and Smith in Reference 23. The gain and phase characteristics of the open-loop transfer function of pitch attitude error, including a specified pilot model, are overlaid on a Nichols chart. The pilot model has a 0.3 sec time delay, plus lead/lag compensation as illustrated in Fig. 5. Pilot gain and equalization are adjusted as necessary to meet the closed-loop bandwidth and droop standards shown in Figure 6. The resulting closed-loop resonance and pilot compensation are then compared to the boundaries indicated in Figure 7, which also contains flying qualities interpretations of the various regions of the figure. Bandwidths were found which resulted in quite good correlation of these boundaries with pilot comments.

Examples of further validation of the Neal-Smith criteria are contained in Ref. 30 for the B-1 bomber and Ref. 31 for an F-4C with a highly augmented command augmentation system. Reference 32a presented further work based on the data of Reference 19 (LAHOS data), and suggested some modifications to the original rules. Radford and Rogers Smith (a) note that landing is a high bandwidth task, (b) propose not forcing 3 db of droop, (c) found a reduced pilot time delay necessary to fit the LAHOS data and (d) suggest the need of an additional, "adaptability" parameter relating variations in needed pilot lead, peak amplitude ratio and bandwidth.

We felt that we would need a better definition of the required bandwidth for each task before this criterion could be used in the general format of MIL-F-8785C. It can certainly be a help in the design process. In this regard, extreme sensitivity of the parameters to small changes in bandwidth is an indication of potential problems.

Reference 23 also discussed a way to simplify or approximate the criterion, which was developed into a proposed revision in Ref. 15. This proposed requirement is a function of only open-loop characteristics of the pitch response, as shown in Figure 8.

Step Target Tracking

In Ref. 33, Onstott proposes a two-stage model of tracking a step change in aim error during target tracking. Both models incorporate a 0.3-second time delay and adjustable lead, and the second model also has an integral term. The model parameters and the switching time are selected to maximize time on target (with a pipper diameter of 5 milliradians). Onstott used the Neal-Smith data to divide the rms error vs. time-on-target plane into regions of flying qualities Levels (Fig. 9). His finding that both quickness of acquisition (small rms pitch error) and time on target determine flying qualities acceptability is obviously correct in general. This is another approach that we feel would be an aid in the design process but is not defined sufficiently to be used as the basis for a specification. Ref. 32a presents some further analysis.

Paper Pilot and Similar Optimal Pilot Models

"Paper Pilot" is now an adult. Anderson proposed this closed-loop flying qualities prediction technique in 1960 (Ref. 34) as a unified way to specify hover dynamics for both rate and attitude control systems. Paper Pilot adjusts parameters of a pilot model appropriate to the task, such as (for hover)

$$Y_{p_\theta} = -K_p(T_{L_\theta} s + 1)\ (s - 2/\tau)/(s + 2/\tau)$$

$$Y_{p_x} = K_{p_x}\ (T_{L_x} + 1)$$

to minimize a task-dependent rating function, which he first took to be

$$R = R_1 + R_2 + R_3 + 1.0$$

$$R_1 = \frac{\sigma + 10\sigma_q - \sigma_m}{\sigma_m},\quad 0 \le R_1 \le 2.5$$

$R_2 = 2.5T_{L_\theta} \leq 3.25$ sec.

$R_3 = T_{L_x} \leq 1.2$ sec.

σ_q In radians/second

σ_m, The required performance, determined empirically to be 0.8 feet

for compensating in the presence of atmospheric turbulence. The parameters are bounded by the limits shown. Several theses extended the model to other piloting tasks. Dillow and Picha (Ref. 35) used a pilot model with a "smarty-pants Kalman filter" in single-and dual-axis tracking tasks with thresholds. They were able to find weighting functions which gave good to excellent correlation between analysis and experiment in hover, pitch tracking and roll tracking. Using these cost functions they obtained good correlation of trends, if not rating and performance, with other experimental data. Dillow and Picha's pilot model uses pilot-perceived control variables and their rates. RMS control rate (adjusted to correspond to 0.1 sec. neuromuscular lag) is a measure of pilot workload, although incomplete understanding of the parameter is professed.

More recent closed-loop analyses utilizing optimal pilot models include the work of Hess (Ref. 36) and Levison (Ref. 32b). Although various investgators have claimed success, particularly with single-axis tracking, much of the flying qualities community remains reluctant to use closed-loop parameters directly in a specification (Ref. 37a). For the present, pilot-vehicle analysis has achieved wider acceptance as a design tool, e.g., Ref. 38, than as a form of design requirement.

Heading Control Criteria

The use of "coordinated" aileron and rudder is accepted as normal piloting technique, provided that the required rudder inputs are not too demanding a task. Reference 14 indicated that the response to rudder inputs necessary to coordinate turns plays a dominant role in evaluations, and proposed a quantitative measure of acceptable and unacceptable characteristics. The analysis of coordination in turn entry (defined as keeping sideslip close to zero, where $T_{\beta_{tc}}$ and $T_{\beta_{ac}}$ are the mid-frequency zeros of the β/δ_{rc} and β/δ_{ac} transfer functions respectively, δ_{rc} and δ_{ac} are yaw and roll pilot's command inputs, and δ_{rc} (t) is the "ideal" δ_{rc} to accompany a step δ_{ac})is based on the following parameters:

PARAMETER	ANALYTICAL FUNCTION	PILOT-CENTERED FUNCTION
$\mu = \frac{T_{\beta_{rc}}}{T_{\beta_{ac}}} - 1 \simeq \frac{\delta_{rc(3\,sec)}}{\delta_{rc(0)}} - 1$	Defines shape of Y_{CF}	Determines complexity of rudder activity necessary for ideally-coordinated turns. Also defines phasing of heading results when rudder is not used.
$N'_{\delta_{as}}/L'_{\delta_{as}}$	Defines magnitude of Y_{CF}	Determines magnitude of rudder required and/or high frequency yawing induced by aileron inputs.

Boundaries developed using data from six sources are presented in Figure 10.

Calspan proposed revisions[15] to ameliorate some of the problems encountered in application of the roll-sideslip coupling requirements of MIL-F-8785/C. By deleting the spiral mode from time history traces, the Dutch roll oscillation would be easier to work with. A further refinement would substitute the bank angle and roll rate at the first peak for the now-specified average value; new boundaries were drawn in terms of these parameters and a different phase angle: of the ϕ or p response, not the sideslip response. We felt these changes to be no more than a marginal improvement, while they would add somewhat to the complications of an already overly complex set of requirements.

It is widely recognized[3, 15, 27, 29] that lateral acceleration is an important parameter of roll response, heading control and turbulence response. An adequate way to account for this factor quantitatively in a generalized flying qualities requirement, however, has not yet been found. The various requirements on dynamic sideslip response give partial but not full coverage.

We are still looking for a simpler yet more meaningful, more comprehensive way to specify lateral-directional dynamics.

The foregoing is only an overview in order to illustrate the diversity of criteria that have been proposed. Other References, such as 32c and two Dutch reports (NLR)[39, 40] present surveys of contemporary flying qualities criteria with checks against flight simulator results for a transport airplane utilizing active control technology.

CURRENT STATUS

As we have indicated, the past decade saw a large number of criteria proposed, frequently substantiated by only one data set. The designers were faced with the problem of deciding which criterion or combination of criteria to use. Evidence of the dilemma is presented by Rickard (Ref. 32c) who indicates that thirteen criteria or combinations of criteria were considered in researching longitudinal flying qualities. The recommendation for large transport configurations was to use a version of the Neal-Smith criterion plus the flight path stability requirement from MIL-F-8785B. Yet more evidence of the designers view of the available criteria is contained in Reference 41. For the AFTI control law development (an F-16 modification for technology demonstration) MIL-F-8785B, C*, Neal-Smith and Onstott's step target tracking criterion were all used in different combinations for different modes. Taken at face value this would imply a serious analytical use of the available criteria. However, "In some cases, what the criteria considered "good" flying qualities did not provide the desired results in the simulator." As discussed by Rogers Smith (Ref. 42) the use of a ground-based simulator to "tune" a design is no guarantee of acceptable flying qualities. The proof (or otherwise) of the AFTI development will be revealed by the forthcoming flight tests.

What is our approach? For the 1980 version of the specification, MIL-F-8785C[28], we have retained the large data base of MIL-F-8785B by using primarily the same modal requirements - but explicitly applied to equivalent system parameters (Figure 11a). In this way we apply the requirements based on modal characteristics to the overall aircraft response. There should now be no implication that we are considering dominant modes. We also believe that this is responsive to the needs of designers. Failure of an equivalent system parameter to meet the requirement then indicates the nature of the problem (e.g., damping, delay or lag). We acknowledge that the use of equivalent systems is not a magic solution to good flying qualities; however, properly used it is a good tool for designing or evaluating advanced configurations which are becoming indiscriminately complex.

In the past, both operational experience and flying qualities research were largely limited to aircraft which behaved in the classical manner: response to control and disturbance inputs characterized by transfer functions of familiar form. The effects of additional dynamics introduced through the flight control system were recognized at the time MIL-F-8785B was written, but limited knowledge prevented adequate treatment. Still, aircraft design developments continue to emphasize equalization to "improve" aircraft response. Certainly one would expect that failure to consider one or more dynamic modes in the frequency range of pilot control would give erroneous results. Prime examples include the F-14[43] and the YF-17[44] designs. The F-14's stability augmentation system was designed to increase the low short-period frequency. At one stage of the design it appeared to do that well in landing approach, but it also introduced higher-order dynamics which resulted in an overall "effective short-period frequency" little changed from augmentation-off. In a flight evaluation of predicted YF-17 characteristics using the FDL-Calspan NT-33 Variable Stability Airplane, pilots rated the short-period response poor to bad. It is pertinent that a configuration intended to have good flying qualities got "good" pilot ratings in flight only after the flight control system compensation had been simplified. Reference 14 cites a number of such problems with recent airplanes.

There are several simple mechanizations which can augment stability without increasing the order of the system response. However, prefilters, forward-loop compensation, crossfeeds, etc. are legitimate design tools which are being used on many current aircraft and indeed seem to be the norm. These artifacts do increase system order and we need to be able to account for their effects in the requirements. Thus, with modern flight control and stability augmentation systems, there has been considerable confusion regarding the "proper" selection of modal parameters such as short-period frequency and damping. Correlation of Level 1 flying qualities requirements with characteristics of the bare airframe is certainly not valid for augmented vehicles. We therefore focus attention on the quality of the actual overall response perceived by the pilot, rather than imply consideration of a dominant mode as may be inferred (however incorrectly) for MIL-F-8785B. In concept, the equivalent system approach is consistent with our belief that the pilot desires a clean, classical, second-order response. It may not always be clear exactly what the appropriate variable is, depending on the particular piloting task. We suggest, however, that the problem is to convince designers to use the new flight control technology to do the "old-fashioned" flying qualities better. This requires resisting the urge to incorporate technology for its own sake; for example, see References 42's discussion of flying qualities problems of recent advanced aircraft. The message, then, is to satisfy the intent of the specification. We have assumed that the preceding points will be discussed in more detail in other papers at this conference. Gibson's paper is one excellent example of translating the intent of MIL-F-8785B into design guidance.

The preceding discussion should not be taken as implying that only small benefits can be expected from flight control technology. On the contrary, multimode control allows tuning the "old-fashioned" flying qualities for each different task. Failure requirements can be satisfied by redundant and reconfigurable flight control systems, control forces can be tailored to avoid both the too light and too heavy extremes, etc. Automation of routine pilot tasks is not new, but it is being expanded into newer areas such as Integrated Flight/Fire Control. Finally, completely new modes of operation are possible with the incorporation of direct force control capabilities. The use of direct sideforce control has been shown to be beneficial for air-to-ground weapon delivery when mechanized as a Wings-Level-Turn mode controlled by the rudder pedals. In this form, such a mode complements the conventional flying qualities characteristics instead of replacing them. It is

also interesting to note that, of the various modes evaluated on the CCV YF-16, the best rating was given to the Maneuver Enhancement mode - the use of blended Direct Lift Control to increase the bandwidth of the conventional pitch response.

The equivalent system approach to flying qualities criteria will be discussed in detail in John Hodgkinson's paper later in this conference. After the preceding discussion, however, it is appropriate to discuss the rationale for using equivalent systems in MIL-F-8785C. One obvious advantage is that it preserves the data base of MIL-F-8785B. It should also satisfy the philosophy of the preceding discussion. The overall aircraft response to pilot input has to meet second-order type requirements. There are still some questions of interpretation to be answered, e.g. Figure 11b. These were the subject of much discussion at the recent Flight Dynamics Laboratory symposium,[45] without resolution.

One often encounters more than one equivalent system giving a good fit. Slight differences in the frequency range used, differences in initial parameter values, or differences in optimization procedure can lead to a multitude of "equivalent" systems. The situation is analogous to the nonuniqueness problem encountered by past researchers in analog matching. Although this may present a dilemma for purposes of identifying a plant, in our experience it has not been a problem for purposes of predicting handling qualities Levels; each "good fit" equivalent system for a given higher-order system has generally led to the same prediction of pitch tracking flying qualities.

We are currently developing better guidance for applying equivalent system parameters to the requirements of MIL-F-8785C, (e.g. Reference 45a).

FUTURE DIRECTIONS

We are in the midst of further revision of the specifications into a MIL-Standard and Handbook. The Standard will be only the skeleton of a detailed airplane specification, with blanks for the requirements. The Handbook will contain all the information needed to fill in the blanks for a particular aircraft mission. This will consist of recommended criteria with substantiation, including the possibility of alternative criteria being recommended for a certain airplane class, task or form of control. As outlined by Weingarten[37b] the Aeronautical Systems Division is in the midst of converting the aeronautical specifications it uses to this new format. Coordination with the other Military services is proceeding, and a widespread changeover to this format is envisioned.

The format of the Handbook will also facilitate making clear what the intent of each requirement is, by formalizing and expanding the information in the current backup document.[8] At our recent symposium, discussing the proposed Standard and Handbook, one industry representative was of the opinion that the new documents should be oriented towards forcing a dialogue between the industry flying qualities and control systems disciplines. The implication was that flying qualities engineers may understand the requirements but they probably do not have control over the "output" of the flight control system. An obvious danger of this situation is that we may end up with "perfect" control of the wrong variable.

At the FDL symposium,[45] there was a feeling that a small group of researchers were discussing minute details of the criteria, whereas practical design guidance was at a premium. A goal of the new MIL-Standard and Handbook is to provide such guidance. One part of this is to provide alternate criteria in the Handbook. As the earlier discussion showed, many criteria have been developed which are valid for at least a particular data set or application. There must be some validity in all of them; an objective in the Handbook will be to emphasize the similarities between the different criteria and where they are most effective. The Handbook will also be developed continuously. The first major activity will be the addition of STOL requirements.

There is a continuing need to define the piloting requirements. As aircraft and mission tasks become more complex certain tasks may be automated, such as Integrated Flight/Fire Control (IFFC). Optimum performance is achieved through the correct balance of manual and automatic control inputs. Operational requirements such as night in-weather ground attack will only be satisfied by a truly integrated design of the aircraft response characteristics, display and controller characteristics and the automatic functions. We plan to add more rigorous consideration of closed-loop or pilot-in-the-loop criteria.

Transfer functions are inherently linear representations of the actual dynamics. Various requirements state specifically that they apply to all amplitudes of motion and to each cycle of an oscillation. Generally, the intent in the specification is to establish bounds on parameters of a rational quasilinear representation of the system for all reasonable amplitudes of control inputs and airplane motions (with separate requirements at such extremes as residual oscillations and stalling). The control saturation due to very high feedback gains can result in poor flying qualities at moderate to large amplitudes, by altering the motion parameters too severely from their values at small amplitude.[14] In more general terms, a large portion of the specification is based on small perturbation analysis. We are initiating an effort to develop non-linear analysis techniques for use in the Handbook.

Lastly, we need to ensure that the new Standard and Handbook will be used in future aircraft system procurements. This requires that they be validated against a recent, highly augmented aircraft. We plan to identify the requirements that are critical to both performance and cost. Piloted simulation will be used to study the sensitivity of critical

requirements, i.e., what is the effect of off-nominal conditions? By implication, the process will identify the requirements that should be weighted heavily vs. ones that do not have much impact. If this can be done, we feel that both ASD and industry will accept the flying qualities Standard and Handbook as essential to preventing the delays and costs of curing flying qualities deficiencies during flight test.

CONCLUSIONS

We have seen that the flying qualities research of the 1970s produced a wealth of criteria. At the same time the use of flight control technology to modify flying qualities expanded and the specification lost much of its credibility among flight control designers. The current solution is to apply the requirements to parameters of an equivalent match to the actual high-order dynamics. This will make clear that the requirements apply to the overall response to pilot or external input, not to any particular or dominant mode. A further revision is currently in progress which is intended to address the mission requirements more directly. For the future we see even more emphasis on closed-loop criteria applied to the piloting task.

REFERENCES

1. Advertisement and Specification for a Heavier-than-air Flying Machine, Signal Corps Specification No. 486, December 23, 1907.

2. Foulois, Maj Gen. B. D., "The Day I Taught Myself to Fly," Airman, September 1965 (reprinted March 1970).

3. Soule, H. A., "Preliminary Investigation of the Flying Qualities of Airplanes," NACA Report 700, 1940.

4. Stability and Control Characteristics of Airplanes, Army Air Forces Specification No. C-1815, 31 August 1943.

5. Phillips, W. H., "Appreciation and Predictions of Flying Qualities," NACA Report 927, 1959.

6. Flying Qualities of Piloted Airplanes, Military Specification MIL-F-8785B (ASG), 7 August 1969.

7. Perkins, C. D., and R. E. Hage, Airplane Performance, Stability and Control, John Wiley & Sons, Inc., New York, 1949.

8. Chalk, C. R., et al., Background Information and User Guide for MIL-F-8785B (ASG), "Military Specification-Flying Qualities of Piloted Airplanes," AFFDL-TR-69-72 (AD860856), 9 August 1969.

9. Shomber, H. A., and Gertsen, W. M., "Longitudinal Handling Qualities Criteria: An Evaluation," AIAA Paper No. 65-780, November 1965.

10. Fortenbaugh, R. L., "Proposed Revisions to MIL-F-8785B Based on Equivalent Airframe Dynamics;" Naval Air Development Center Working Paper, January 1974.

11. A'Harrah, R. C., and J. L. Lockenour, "Approach Flying Qualities - Another Chapter," AIAA Paper No. 69-895, August 1969.

12. Aeroplane Design Requirements, AvP 970, Royal Aircraft Establishment, UK, August 1977.

13. Bihrle, W. Jr., "A Handling Qualities Theory for Precise Flight Path Control," AFFDL-TR-65-198, June 1966.

14. A'Harrah, R. C., and R. J. Woodcock, "The Military Flying Qualities Specification, A Help or a Hindrance to Good Fighter Design?" Combat Aircraft Maneuverability, AGARD-CP-319.

15. Chalk, C. R., et al., "Revisions to MIL-F-8785B (ASG) Proposed by Cornell Aeronautical Laboratory under Contract F33615-71-C-1254," AFFDL-TR-72-41, Air Force Flight Dynamics Laboratory, Wright-Patterson AFB, Ohio, April 1973.

16. Bishoff, D. E., "Longitudinal Equivalent Systems Analysis of Navy Tactical Aircraft," AIAA Paper No. 81-1775, August 1981.

17. Hoh, R. H., et al., "Proposed MIL Standard-Handling Qualities of Piloted Airplanes," forthcoming AFWAL TR.

18. Ashkenas, I. L., et al., "Recommended Revisions to Selected Portions of MIL-F-8785B (ASG) and Background Data," AFFDL-TR-73-76, Air Force Flight Dynamics Laboratory, Wright-Patterson AFB, Ohio, August 1973.

19. Smith, R. E., "Effects of Control System Dynamics on Fighter Approach and Landing Longitudinal Flying Qualities," AFFDL-TR-78-122, March 1978.

20. Smith, R. H., and N. D. Geddes, "Handling Quality Requirements for Advanced Aircraft Design - Longitudinal Mode," AFFDL-TR-78-154, August 1979.

21. Malcolm, L. G., and Tobie, H. N., "New Short Period Handling Quality Criterion for Fighter Aircraft," Boeing Report D6-17841T/N, September 1965.

22. Kisslinger, R. L., and Wendl, M. J., "Survivable Flight Control System Interim Report No. 1 Studies, Analysis and Approach, Supplement for Control Criteria Studies," AFFDL-TR-71-20, May 1971.

23. Neal, T. P., and Smith, R. E., "An In-Flight Investigation to Develop Control System Design Criteria for Fighter Airplanes," AFFDL-TR-70-74, Vols I and II, December 1970.

24. Abrams, C. R., "A Performance Index for Response Evaluation of Highly Augmented Military Aircraft," NADC-AM-7103, Naval Air Development Center, Pa., October 1971.

25. Brulle, R. V., and Moran, W. A., "Dynamic Flying Qualities Investigation," AFFDL-TR-74-142, October 1974.

26. Di Franco, D. A., "Flight Investigation of Longitudinal Short-Period Frequency Requirements and PIO Tendencies," AFFDL-TR-66-2, February 1966.

27. Chalk, C. R., "Calspan Recommendations for SCR Flying Qualities Design Criteria," Calspan Report No. 6241-F-5, April 1980.

28. Flying Qualities of Piloted Airplanes, Military Specification MIL-F-8785C, November 1980.

29. Weingarten, N. C., and C. R. Chalk, "In-Flight Investigation of Large Airplane Flying Qualities for Approach and Landing," AFWAL-TR-81-3118, September 1981 (forthcoming).

30. Gerken, G., "Comparison of the B-1 to Three Longitudinal Maneuver Criteria," ASD-TR-75-29, Aeronautical Systems Division, Wright-Patterson AFB, Ohio, 1975.

31. Kogler, R. H., "A Limited Analysis of an F-4C Aircraft with a High-Gain Control Augmentation System and Comparison with Flight Test Data," ASD-TR-73-21, May 1973.

32. Flying Qualities Design Criteria, Proceedings of AFFDL Flying Qualities Symposium held at Wright-Patterson AFB in October 1979, AFWAL-TR-80-3067, May 1980.

    a. Radford, R. C., and R. E. Smith, "Approach and Landing Flying Qualities Requirements," pp. 78-126.

    b. Levison, W. H., "A Model-Based Technique for Predicting Pilot Opinion Ratings for Large Commercial Transports," pp. 43-77.

    c. Rickard, W. W., and D. S. Joshi, "Transport Aircraft Flying Qualities Research," pp. 31-42.

33. Onstott, E. D., and Faulkner, W. H., "Prediction, Evaluation and Specification of Closed Loop and Multiaxis Flying Qualities," AFFDL-TR-78-3, February 1978.

34. Anderson, Ronald O., "A New Approach to the Specification and Evaluation of Flying Qualities," AFFDL-TR-69-120, June 1970.

35. Dillow, Maj James D., and Picha, 1st Lt Douglas G., "Application of the Optimal Pilot Model to the Analysis of Aircraft Handling Qualities," AFIT-TR-75-4, August 1975.

36. Hess, R. A., "A Method for Generating Numerical Pilot Opinion Ratings Using the Optimal Pilot Model," NASA TM X-73101, February 1976.

37. Proceedings of AFFDL Flying Qualities Symposium Held at Wright State University 12-15 September 1978, AFFDL-TR-78-171, December 1978:

    a. George, F. L., "Closed-Loop Flying Qualities Criteria," pp 574-586.

    b. Weingarten, J. L., "Aeronautical Procurement - The Primary Specification System," pp. 499-512.

38. Schmidt, D. K., "Pilot-Optimal Augmentation of the Air-to-Air Tracking Task," AIAA Paper No. 79-1747.

39. Mooij, H. A., et al., "Determination of Low-Speed Longitudinal Maneuvering Criteria for Transport Aircraft with Advanced Flight Control Systems," NLR TR 79127U, The Netherlands, December 1979.

40. Van Gool, M. F. C., "Roll Response Criteria for Transport Aircraft with Advanced Flight Control Systems in the Landing Approach and Touchdown," NLR TR 80103L, The Netherlands.

41. Barfield, F. A., and B. W. VanVliet, "AFTI/F-16 Advanced Multimode Control System Design for Task-Tailored Operation," AIAA-81-1707, August 11-13, 1981.

42. Smith, Rogers E., "Evaluating the Flying Qualities of Today's Fighter Aircraft," Combat Aircraft Manoeuvrability, AGARD-CP-319 (Supplement) classified NATO Confidential, Flight Mechanics Panel Symposium held in Florence, Italy, 5-8 October 1981.

43. A'Harrah, R. C., "Flight Simulation - A Significant Aid in Aircraft Design," AGARD Conference Proceedings No. 119, Stability and Control, Braunschweig, Germany, 10-13 April 1972.

44. Hall, G. W., and Harper, R. P., Jr., "In-Flight Simulation of the Light Weight Fighters," AIAA Paper No. 75-986, August 1975.

45. Design Criteria for the Future of Flight Controls, Flight Dynamics Laboratory Symposium held in Dayton, Ohio, March 1982 (Proceedings to be published as an AFWAL TR).

    a. Gentry, T. A., "Guidance for the Use of Equivalent Systems with MIL-F-8785C."

46. Hodgkinson, J., et al., "Equivalent Systems Approaches to Handling Qualities Analysis and Design Problems of Augmented Aircraft," AIAA Paper No. 77-1122, August 1977.

47. A'Harrah, R. C., W. J. Lamanna and J. Hodgkinson, "Are Today's Specifications Appropriate for Tomorrow's Airplanes?" Stability and Control, AGARD Conference Proceedings No. 260, September 1978.

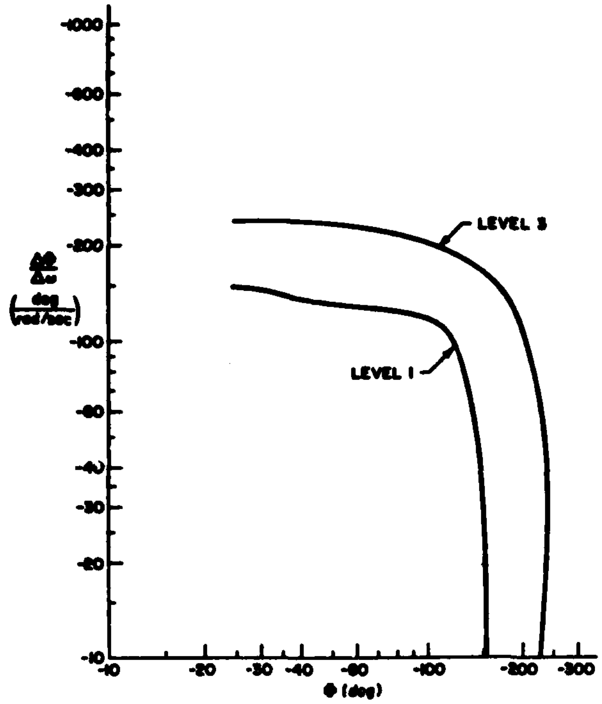
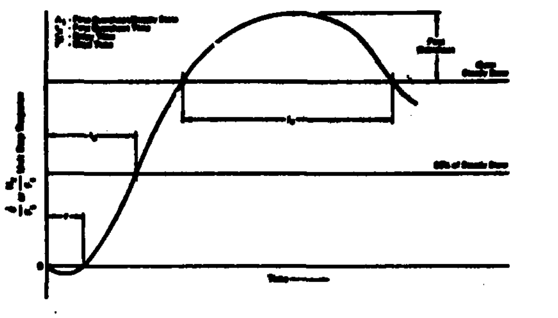
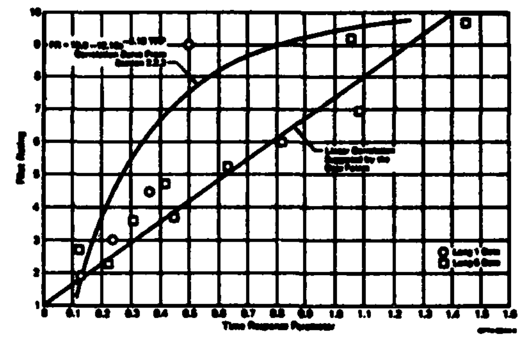


Figure 1. Longitudinal Attitude Control - Category C Requirements (Ref 18)



a) TRP Definitions



b) Pilot Rating Variation with TRP

Figure 3. Definition and Correlation of Time Response Parameter (Ref 24)

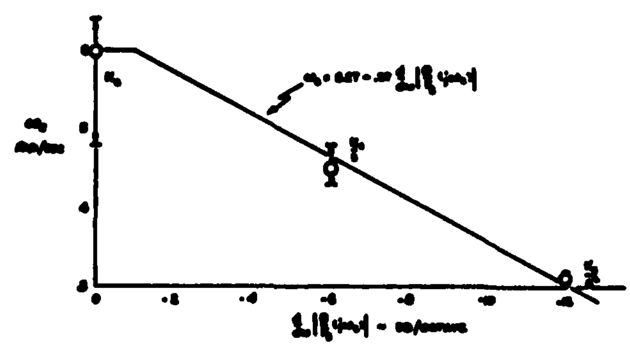


Figure 2. Specification of the Criterion Frequency (Ref 20)

Transient Peak Ratio

The transient peak ratio $\Delta q_2/\Delta q_1$ shall be equal to or less than the following:

Level	$\Delta q_2/\Delta q_1$
1	.30
2	.60
3	.85

Rise Time Parameter

The parameter $\Delta t = t_2 - t_1$ shall have a value between the following limits:

	Nonterminal Flight Phases		Terminal Flight Phases
Level	Min Δt Max	Level	Min Δt Max
1	$\underset{\frac{2.74}{V_T}}{(9)} \leq \Delta t \leq \underset{\frac{152.4}{V_T}}{(500)}$	1	$\underset{\frac{2.74}{V_T}}{(9)} \leq \Delta t \leq \underset{\frac{61}{V_T}}{(200)}$
2	$\underset{\frac{.975}{V_T}}{(3.2)} \leq \Delta t \leq \underset{\frac{609.7}{V_T}}{(2000)}$	2	$\underset{\frac{.975}{V_T}}{(3.2)} \leq \Delta t \leq$ [illegible]

where: V_T = ms^{-1} true airspeed.
Constant in parenthesis is used for V_T = ft/sec.

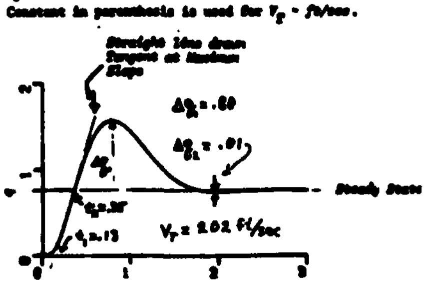


Figure 4. Requirements for Pitch Rate, Response to Step Input of Pitch Controller Force (Ref 27)

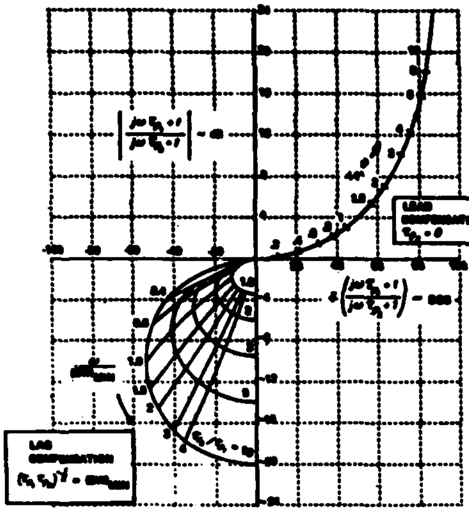


Figure 5. Amplitude - Phase Curves for "Optimum" Pilot Compensation (Ref 23)

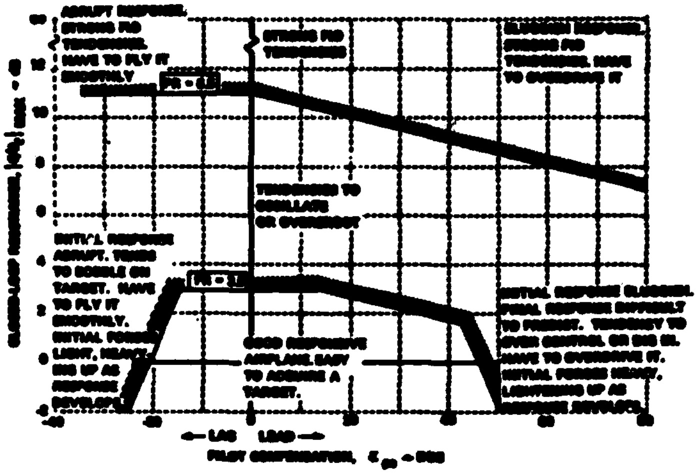


Figure 7. Proposed Criterion for Fighter Maneuvering Dynamics (Ref 23)

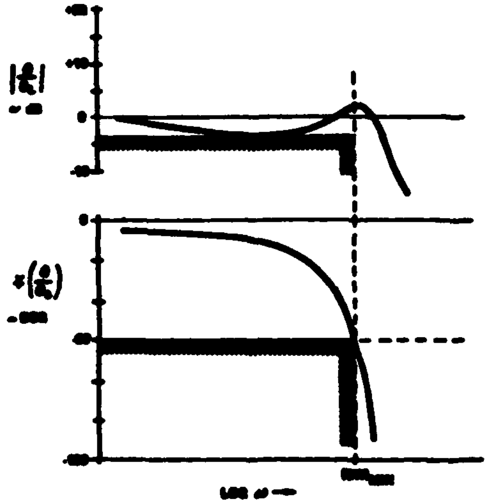


Figure 6. Tracking Performance Standards (Ref 23)

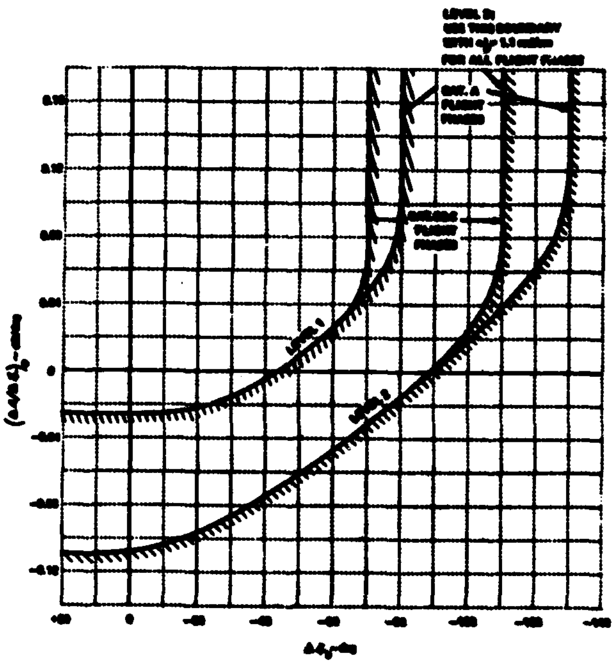


Figure 8. Pitch Maneuver Response Requirements (Ref 15)

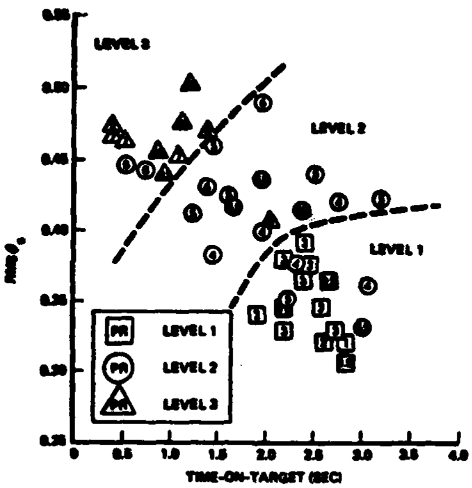


Figure 9. Pilot Ratings vs RMS θ_e and Time on Target (Ref 33)

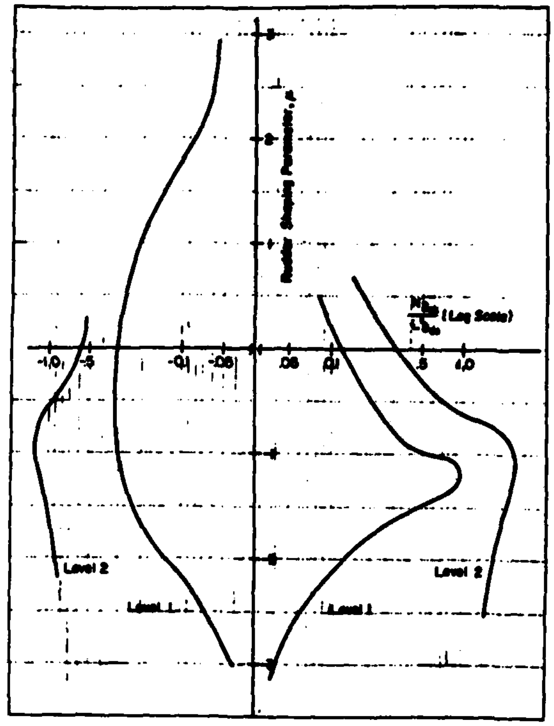
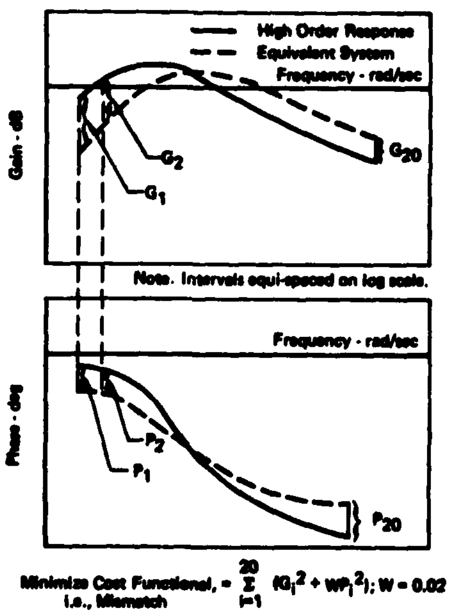
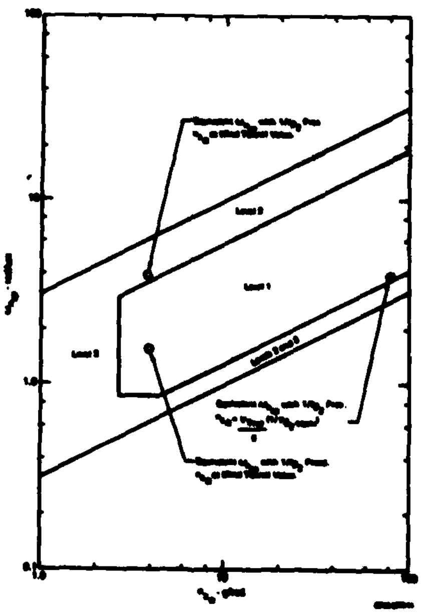


Figure 10. Aileron - Rudder Coordination Limits (Ref 18)



a) Definition



b) Interpretation

Figure 11. Demonstrating Compliance with Equivalent System Parameters (Ref 47)

STATUS OF VTOL AND VSTOL FLYING QUALITIES CRITERIA DEVELOPMENT
WHERE ARE WE AND WHERE ARE WE GOING?

by
John W. Clark, Jr. and Kevin W. Goldstein
Aerospace Engineers
Aircraft and Crew Systems Technology Directorate
US Naval Air Development Center
Warminster, Pennsylvania, USA 18974

SUMMARY

Over the past decade, a number of weaknesses and omissions have been uncovered in the VSTOL and Helicopter Flying Qualities Specifications (MIL-F-83300 and MIL-H-8501A). Identification of these weaknesses has spawned technology development in a number of areas. This paper presents results (both interim and final) in some of these areas, the status of existing data bases and the future criteria development needs as perceived by the US Navy. Specific areas addressed include: (1) information display and IMC (Instrument Meteorological Conditions) flight requirements; (2) criteria definition for highly augmented, multi-mode control schemes; (3) requirements unique to the small seaborne platform operational environment; (4) requirements unique to varied rotor configurations. Both fixed-wing and rotary-wing criteria are considered.

INTRODUCTION

The US VSTOL Flying Qualities Specification, MIL-F-83300 (reference 1), was adopted in December, 1970. Since its adoption, the specification has never been used in the procurement of a new airframe (either fixed-wing or rotary-wing). It has, however, been evaluated through ex post facto application to the characteristics of existing prototype and production fixed-wing VSTOL configurations including the AV-8A, YAV-8A and VAK-191B (references 2 through 4). As a result of these applications, a number of potential weaknesses and shortcomings have been identified in the specification. Both quantitative requirements and qualitative classifications have been found to be in need of revision, primarily in the areas of IMC operation, display requirements, hover and low-speed flight control power and response, highly augmented vehicle dynamic response and small deck shipboard operations. Considerable criteria development effort has been expended over the past five years to improve the requirements in these areas. This paper will attempt to provide an overview of resulting proposed criteria for hover and low-speed flight as well as interim results for the transition/conversion regime for fixed-wing configurations.

With the development of a new generation of rotary-wing aircraft for military operations, it has become apparent that the present helicopter handling qualities specification, MIL-H-8501A (reference 5), cannot accurately assess the characteristics of these aircraft. The fact that MIL-H-8501A was last updated 20 years ago only tends to aggravate the problem. The US Navy Light Airborne Multipurpose System (LAMPS) SH-60B, the US Army Utility Tactical Transport Aircraft System (UTTAS) UH-60A, and the Advanced Attack Helicopter (AAH) all use advanced flight control systems for stability and control augmentation. The need to specify the flying qualities of these state of the art vehicles/control systems has necessitated the use of "type specifications" or "prime item development specifications" uniquely devised for each new aircraft/control system. Many papers have been written describing the numerous shortcomings of MIL-H-8501A in realistically regulating handling qualities of present and future helicopters (references 6 through 10), indicating a very real need for an updated version of the specification. These areas include mission oriented criteria and quantitative criteria addressing degraded flying qualities. To facilitate the development of revised criteria it is necessary first to compile a data base of past and present helicopter stability and control characteristics. This paper presents the beginning of such a compilation.

The SH-60B and CH-53D single rotor helicopters are comparatively analyzed against the fundamental flying qualities characteristics addressed by MIL-H-8501A. Vertical control response and autorotation criteria are not included at this time. Flight test data for the XH-59A Advancing Blade Concept (ABC), the XV-15 tilt rotor, and the CH-46A tandem rotor are also included and discussed.

In the development of the present day VSTOL handling qualities specifications, MIL-F-83300 (reference 1) and AGARD 577 (reference 11), extensive rotary-wing pilot rating data were analyzed to substantiate the finalized hover/low-speed criteria. Although AGARD 577 is not intended to be a helicopter specification and MIL-F-83300 has not been used by the Army or Navy for a helicopter development program, these specifications do supply alternative methods of addressing VTOL handling qualities characteristics. The alternative criteria from MIL-F-83300 and AGARD 577 are directly compared with the criteria from MIL-H-8501A to highlight the helicopter specification deficiencies and vehicle anomalies.

The body of this paper is intended to summarize the primary areas of weakness in applying MIL-F-83300 to fixed-wing configurations and MIL-H-8501A to rotary-wing configurations. Progress made in overcoming these weaknesses is detailed and plans for

future development efforts are presented. Although there is overlap in a number of areas, an attempt has been made to separate fixed and rotary-wing developments for the purposes of discussion.

MIL-F-83300 DEFICIENCIES

As previously stated, experience since the adoption of MIL-F-83300 has identified deficiencies in the specification. A comprehensive review of the specification and identification of its potential weaknesses was performed by Hoh and Ashkenas of Systems Technology, Inc. (reference 12). The significant results of this review are summarized in the following.

As currently written, the specification assumes that "IFR capability is inherent in all military aircraft operational missions and, therefore, the detailed requirements are intended to reflect this assumption". The Navy's mission is considerably more demanding than that implied in the specification background. It encompasses operations aboard small seaborne platforms in a highly dynamic environment (up to Sea State 5) and extremely low visibility conditions (700 ft visibility). Since required flying qualities for VSTOL operation in such an environment can be significantly different than those for VMC and other IMC conditions, it is imperative that a specific delineation of task, environment and information display needs be included when defining requirements.

The MIL-F-83300 requirement on cockpit control gradient requires a smooth and stable variation of control force with airspeed for both pitch and roll controllers. While control force gradients of some type are required, it is questionable as to whether gradients with speed are always desirable - particularly when attitude augmentation is implemented. For example, an attitude command control system requires a force gradient with attitude which is most likely independent of airspeed. Therefore, the cockpit control gradient criteria should be dependent on the type of control augmentation implemented. Likewise, cockpit controllers other than center stick type (e.g., sidestick controllers) must be considered for advanced VSTOL aircraft.

Dynamic response requirements in the specification (for both hover/low-speed and forward flight) are based on simulator and flight experience which are now more than 10 years old. For the most part, the requirements are not realistic for the Navy's proposed mission or state of the art control augmentation schemes which exhibit responses which are significantly different from the "classical" VSTOL dynamics considered in the development of MIL-F-83300. Revised requirements are needed which specifically account for characteristics unique to attitude, attitude rate and translational rate augmentation systems.

MIL-F-83300 specifies required vehicle control power in a general fashion in terms of attitude change required in one second or less following an abrupt application of control. With this approach, the breakdown of control power required for trim, maneuvering and disturbance regulation is not addressed. The "attitude in one second" specification is far too general to adequately design highly augmented attitude systems and is especially inadequate for direct force driven translational rate control systems. One proposed solution to the problem is the specification of bandwidth limits on specifically defined equivalent response characteristics. Further analysis, simulation and flight experience is needed in this area.

Height control/response characteristics are currently specified in terms of available control power (both incremental vertical acceleration and steady-state thrust-to-weight) allowable control command lags and minimum rate of climb response. The fact that these parameters are specified independently does not allow for tradeoffs to meet the overall response requirement for the Navy shipboard landing task. It has been proposed that specifying a critical maneuver in a given disturbance environment will allow for design trades between available thrust level, engine response time and inherent aircraft/control dynamics (e.g., equivalent height damping) to achieve the desired overall response characteristics.

In general, the forward flight (35 knots to conversion speed) and transition requirements of MIL-F-83300 are inconsistent with the corresponding requirements of the CTOL specification, MIL-F-8785C (reference 13). Not only is there inconsistency in the quantitative level of the requirements but also in the format in which they are specified. MIL-F-8785C now uses "equivalent" dynamics where dominant mode dynamics were previously specified. An update of the appropriate sections of MIL-F-83300 is required to be consistent with the equivalent dynamics approach, particularly at the conversion speed interface.

As indicated by the preceeding summary, a number of sections of MIL-F-83300 appear to be in need of some degree of revision. Table 1 lists the specification paragraphs which have been identified and indicates those areas where effort has been, is being or is planned to be devoted to each by US Navy sponsored research, and whether or not a revision has been proposed.

The following sections of this paper will detail the results of studies to date which have culminated in proposed criteria in two areas - (1) Control/Display System Criteria and (2) Hover/Low-Speed Equivalent System Criteria. These sections are followed by interim results of ongoing analysis looking at Forward Flight/Transition

TABLE 1: SUMMARY OF CRITERIA REVISION EFFORTS

PARAGRAPH	SUBJECT	Recent Research	Ongoing Research	Planned Research	Proposed Revision
1.2	Application				
1.2.2	Operation under instrument flight conditions......	X	O	O	X
3.2	Hover and low speed				
3.2.1	Equilibrium characteristics				
3.2.1.3	Cockpit control gradients......................	X	X	O	O
3.2.2	Dynamic response requirements				
3.2.2.1	Pitch (roll)...................................	X	O	O	X
3.2.2.2	Directional damping............................	X	O	O	X
3.2.3	Control characteristics				
3.2.3.1	Control power..................................	X	X	O	O
3.2.3.2	Response to control input......................	X	O	O	X
3.2.3.3	Maneuvering control margin.....................	X	O	O	O
3.2.4	Control lags...................................	X	O	O	X
3.2.5	Vertical flight characteristics				
3.2.5.1	Height control power...........................	X	O	O	O
3.2.5.2	Thrust magnitude control lags..................	X	O	X	O
3.2.5.3	Response to thrust magnitude control input......	X	O	X	O
3.3	Forward flight				
3.3.1	Longitudinal equilibrium........................	X	X	O	O
3.3.2	Longitudinal dynamic response...................	X	X	O	O
3.3.5	Pitch control effectiveness in maneuvering flight				
3.3.5.1	Maneuvering control margins.....................	O	O	X	O
3.3.5.2	Speed and flight-path control...................	O	X	X	O
3.3.7	Lateral-directional characteristics				
3.3.7.1	Lateral-directional oscillations (Dutch Roll)...	O	O	X	O
3.3.7.2	Roll mode time constant.........................	O	O	X	O
3.3.7.3	Spiral stability................................	O	O	X	O
3.3.8	Roll-sideslip coupling..........................	O	O	X	O
3.3.8.1	Bank angle oscillations.........................	O	O	X	O
3.3.8.2	Sideslip excursions.............................	O	O	X	O
3.3.8.4	Turn coordination...............................	O	O	X	O
3.4	Transition				
3.4.1	Acceleration-deceleration characteristics.........	O	X	X	O
3.4.2	Flexibility of operations........................	O	X	X	O
3.4.3	Tolerance in transition program..................	O	X	X	O
3.4.4	Control margin...................................	O	O	X	O
3.5	Characteristics of the flight control system				
3.5.1	Mechanical characteristics				
3.5.1.1	Control centering and breakout forces...........	O	O	X	O
3.5.1.2	Cockpit control force gradients.................	O	O	X	O
3.7	Atmospheric disturbances...........................	X	O	O	X

Equivalent System Criteria.

CONTROL/DISPLAY SYSTEM CRITERIA

The complex interaction between displayed information and control augmentation available to the pilot has been known and studied over the years for CTOL aircraft. Research has concentrated on air-to-air tracking and, to some extent, on limited visibility, conventional approach and landing. The Navy VSTOL mission requires operational capability which includes extreme low visibility recovery aboard small ships in up to Sea State 5 conditions (reference 14). If this goal is to be achieved, the display systems (both Head Up (HUD) and Head Down (HDD)) must become an active, integrated part of the aircraft control system design.

The trade between increasing display sophistication and increasing control complexity was hypothesized in a 1972 AGARD Report (reference 15) and is repeated here in Figure 1. As depicted in the figure, system capability can be increased by either "sophisticating" the display (increasing information/integration) or "sophisticating" the control system (successively augmenting outer loops). Inherent in this hypothesis is increased system cost regardless of the approach taken. Note that the axes of Figure 1 are not quantified in any way and, therefore, give little specific guidance to the system designer.

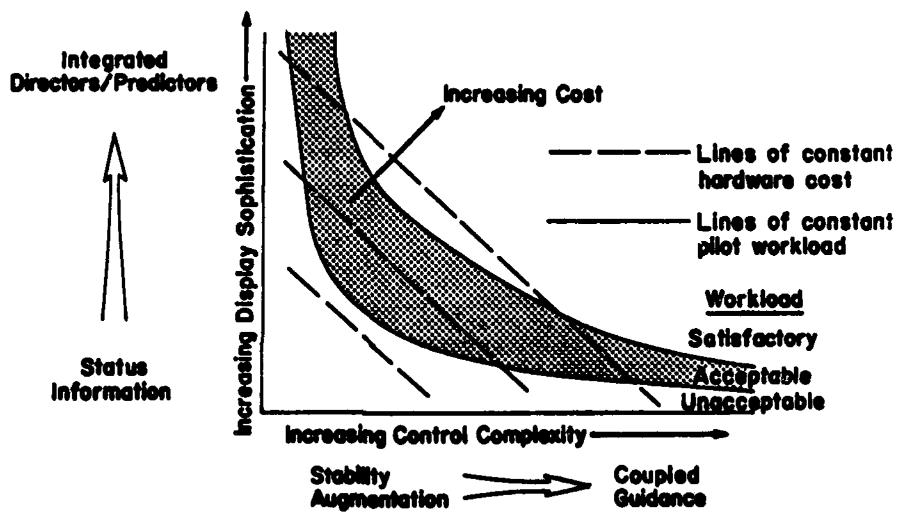


FIGURE 1: AGARD CONTROL/DISPLAY TRADE-OFF

Two recent research programs conducted by Calspan Corporation using the Navy's X-22A Variable Stability VSTOL Research Aircraft attempted to quantify the control/display tradeoff. The first of these studied decelerating, descending transition to hover under IMC conditions utilizing an HDD (see reference 16 for details). Control complexity varied from a basic rate augmentation system to decoupled velocity control. The display formats were varied from basic position situation information to integrated flight director information. Figure 2 summarizes the results. The general conclusion of the study, as evidenced by Figure 2, is that the hypothesized interaction between control augmentation and display content is exhibited in flight. The following specific conclusions also resulted:

> The minimal level of displayed information must include translational velocity information to obtain acceptable performance, regardless of the level of control augmentation. This requirement is primarily hover oriented and reflects the pilot's dislike of having to obtain translational rates implicitly from the movement of symbols on the display.
>
> Rate augmentation alone is not acceptable for the total IMC task unless full control director information is provided. Performance with the rate system in crosswinds became unacceptable even with full director information.
>
> Decoupling and augmenting the longitudinal and vertical velocity responses to control inputs considerably enhanced task performance and tended to eliminate the variation of pilot rating with display sophistication in the configurations where ground velocity was explicitly displayed.

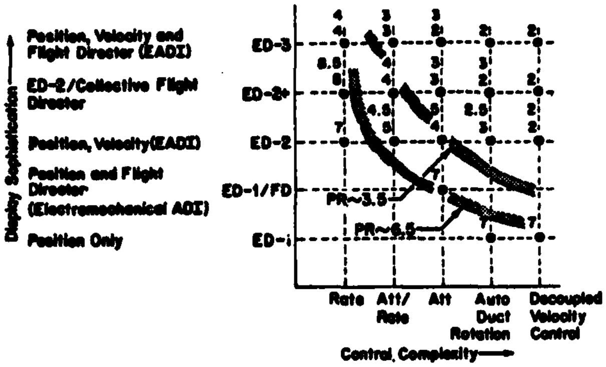


FIGURE 2: X-22A TASK IV FLIGHT RESEARCH RESULTS

The second program (reference 17) investigated control/display tradeoffs for a vectored-thrust, jet lift VSTOL aircraft performing a "one-step" decelerating, descending transition to the hover. The primary conclusions of the previous study were supported by the results of this study.

In general, it has been determined that varying the display content once the display hardware has been fixed results in insignificant change in total system cost. Therefore, once the required level of control augmentation is defined based on mission operating environment, the display content and format may be optimized based on anticipated visibility minimums. Hoh and Ashkenas (reference 18) have proposed a scheme to facilitate the specification of control augmentation level and general display format as a function of visibility level. To better define visibility level, an Outside Visual Cue (OVC) scale has been established which provides a finer delineation than strictly VMC and IMC. The OVC level is defined as a function of the relative availability of position, attitude and velocity visual cues (Figure 3). Using the OVC scale, Table 2 specifies the minimum acceptable level of hover control augmentation for a given display format/content and visibility level. Table 2 is amply supported by available data and appears to be a reasonable approach to specifying VSTOL design guidelines, particularly for hover and low-speed flight. Keep in mind that with this approach the displayed information must be optimized for the given augmentation level and display format.

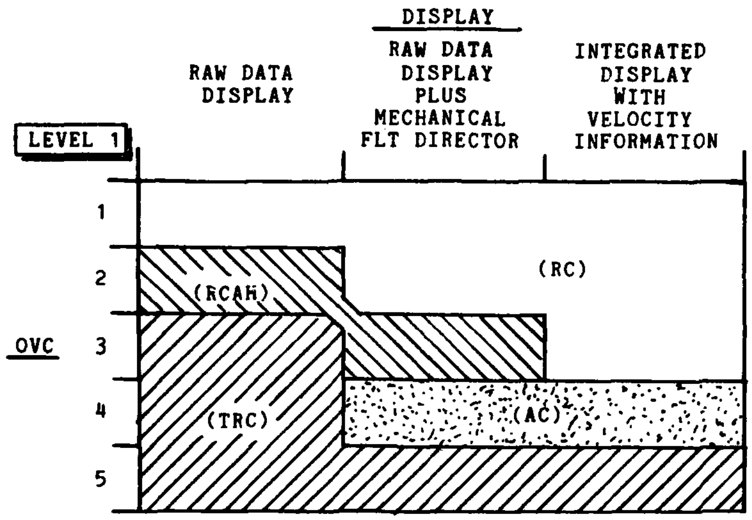
	OVC LEVEL	CUE AVAILABILITY ATTITUDE	POSITION/VELOCITY
VMC	1	Easily obtained	Easily obtained
	2	Full concentration is required to obtain continuous attitude information	Easily obtained
	3	Inadequate in portions of the visual field	Position is obtainable, velocity information is marginal.
	4	Inadequate over most of visual field	Position information is marginal, velocity information is unavailable at times
IMC	5	Unavailable	Unavailable

FIGURE 3: PROPOSED OUTSIDE VISUAL CUE (OVC) SCALE

HOVER/LOW-SPEED EQUIVALENT SYSTEM CRITERIA

Another area of proposed revision is the application of low-order dynamic criteria to complex, high-order closed-loop aircraft/control dynamics of the type most likely to be exhibited by advanced VSTOL aircraft. Current thinking within the Navy and elsewhere tends to support the "Equivalent System" (ES) approach implemented by Hodgkinson and LaManna (reference 19) and included in the revision to the CTOL Flying Qualities Specification, MIL-F-8785C (reference 13). A recent Navy sponsored study (reference 20) developed guidelines for the application of ES analysis to hover and low-speed flight and revised dynamic criteria in an ES format have been proposed.

TABLE 2: MINIMUM LEVEL OF HOVER CONTROL AUGMENTATION



Control Augmentation Hierarchy

Rate Command (RC)

↓

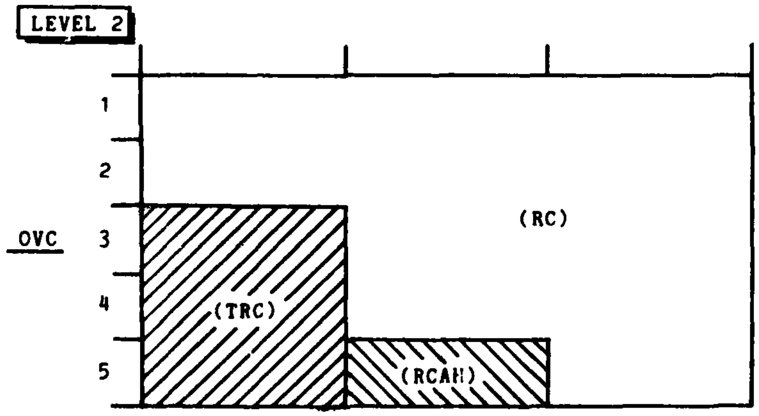
Rate Command / Attitude Hold (RCAH)

↓

Attitude Command (AC)

↓

Translational Rate Command (TRC)



Highly augmented VSTOL configuration models with overall system response types such as those of Table 2 must be simplified in some unified manner so that their characteristics may be readily identified by a limited number of equivalent parameters. The frequency response matching ES technique is being utilized in the CTOL regime with reasonable success and shows promise for similar application to VSTOL configurations. Briefly, the approach used is to match the frequency response (amplitude and phase) of the high order system (HOS) over a given frequency range with a preselected low-order system (LOS) model which minimizes the cost function or mismatch, M, defined by equation (1).

$$M = 20/n \sum((\text{gain(HOS)}-\text{gain(LOS)})^2+57.3(\text{phase(HOS)}-\text{phase(LOS)})^2) \tag{1}$$

where gain is in db and phase is in radians. Large amounts of high frequency lag are accounted for by including a transport lag (delay) in the equivalent system model.

Figure 4 shows the results of applying this procedure to the hover roll attitude command transfer function of the VAK-191B. In this case the HOS is 1st/5th order and the LOS is 0th/2nd order with a transport lag. The figure shows that, for this case, the HOS is matched very well (M=2.1) by the specified LOS form. Note that in order to obtain a good high frequency match, a transport lag of 0.092 second was required in the LOS model. This case is a perfect example of the potentially erroneous system dynamic information which might be gained by considering only the dominant oscillatory mode of the HOS. Here the dominant (only) oscillatory mode of the HOS has a damping and frequency of 0.89 and 8.59 radians/second, respectively; whereas, the corresponding parameters of the oscillatory mode of the LOS are 0.89 and 3.75. The time domain responses are essentially identical (except for the first 0.1 seconds). Applying a frequency domain criteria based on a second order type response to the HOS dominant oscillatory mode would most probably be in error.

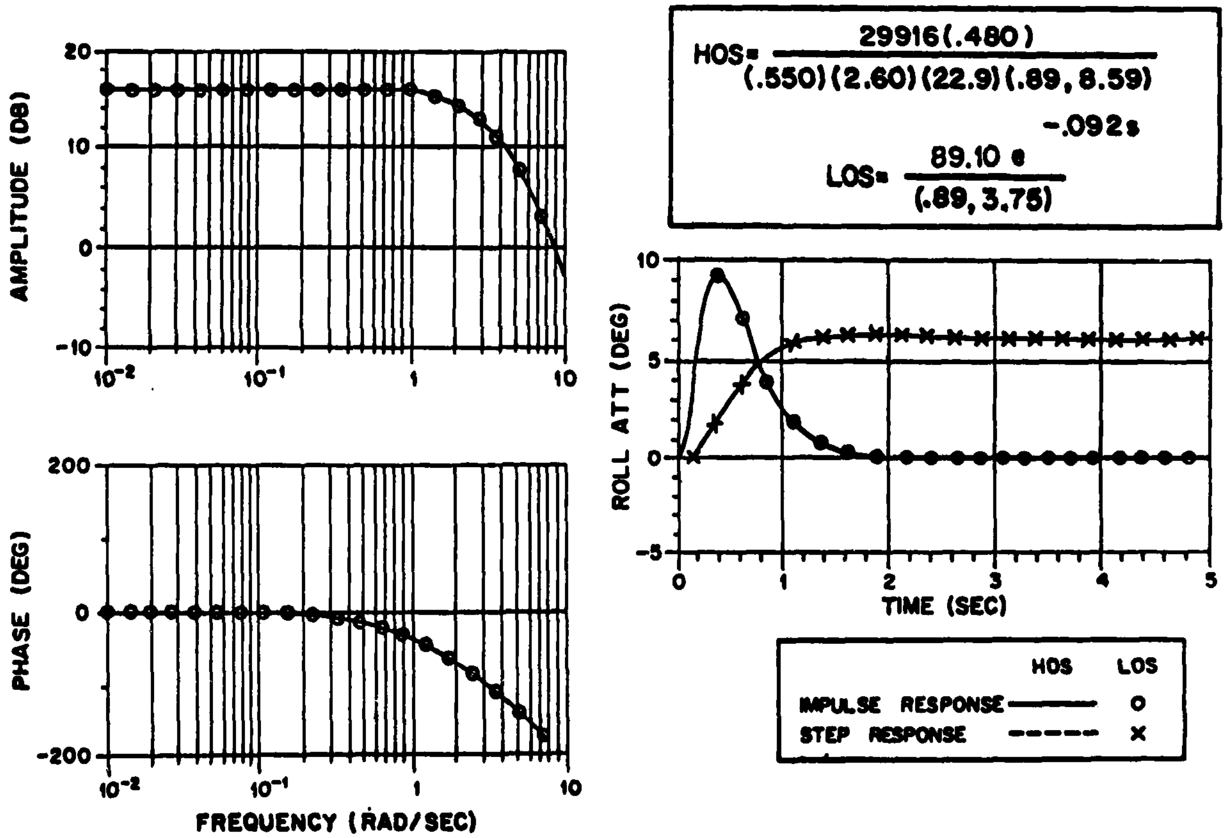


FIGURE 4: VAK-191B ROLL DYNAMICS IN HOVER

Simulation results (reference 20) have indicated that, for attitude control in hovering flight, the equivalent system match should be achieved between 0.5 and 4.0 radians/second to within a mismatch, M, of 100 or less. Equivalent time delay of 0.1 second or less was found to be satisfactory (Level 1) and 0.3 second or less was acceptable (Level 2). Additional simulation has been performed in this area specifically addressing translational rate control in hover and attitude control in forward flight.

Criteria revisions have been proposed which would replace all or part of Paragraphs 3.2.2, 3.2.3 and 3.2.4 of MIL-F-83300. These paragraphs address hover attitude

stability and control dynamics in terms of dominant modal parameters (frequency and damping) and response in one second with no differentiation between various types of augmented response characteristics (rate, attitude, translational rate, etc.). Table 3 summarizes the existing specification Level 1 requirements. The proposed revisions to these requirements are based on an ES approach which allows for several unique LOS formats including attitude, attitude rate and translational rate systems. The criteria for each type of response as well as a proposed classification scheme are detailed in the following.

The proposed LOS form for both attitude and attitude rate augmented vehicle response is given by equation (2).

$$\frac{\text{attitude change}}{\text{cockpit control deflection}} = \frac{K(s+1/T)e^{-\tau s}}{(s+\lambda)(s^2+2\zeta\omega s+\omega^2)} \qquad (2)$$

For attitude response, the LOS form reduces to that of equation (3).

$$\frac{\text{attitude change}}{\text{cockpit control deflection}} = \frac{Ke^{-\tau s}}{(s^2+2\zeta\omega s+\omega^2)} \qquad (3)$$

These forms are presented in reference 18 and are based on an extensive compilation of VSTOL vehicle dynamics. Criteria are defined for the LOS parameters (K, 1/T, etc.) depending upon whether the response is attitude or rate in character. The delination between the two general types of response is provided by the time domain criterion of Figure 5. If the impulse response of the system lies within the boundaries of Figure 5, it must satisfy the attitude criteria. If it violates the boundaries at any point, it is considered to be a rate system.

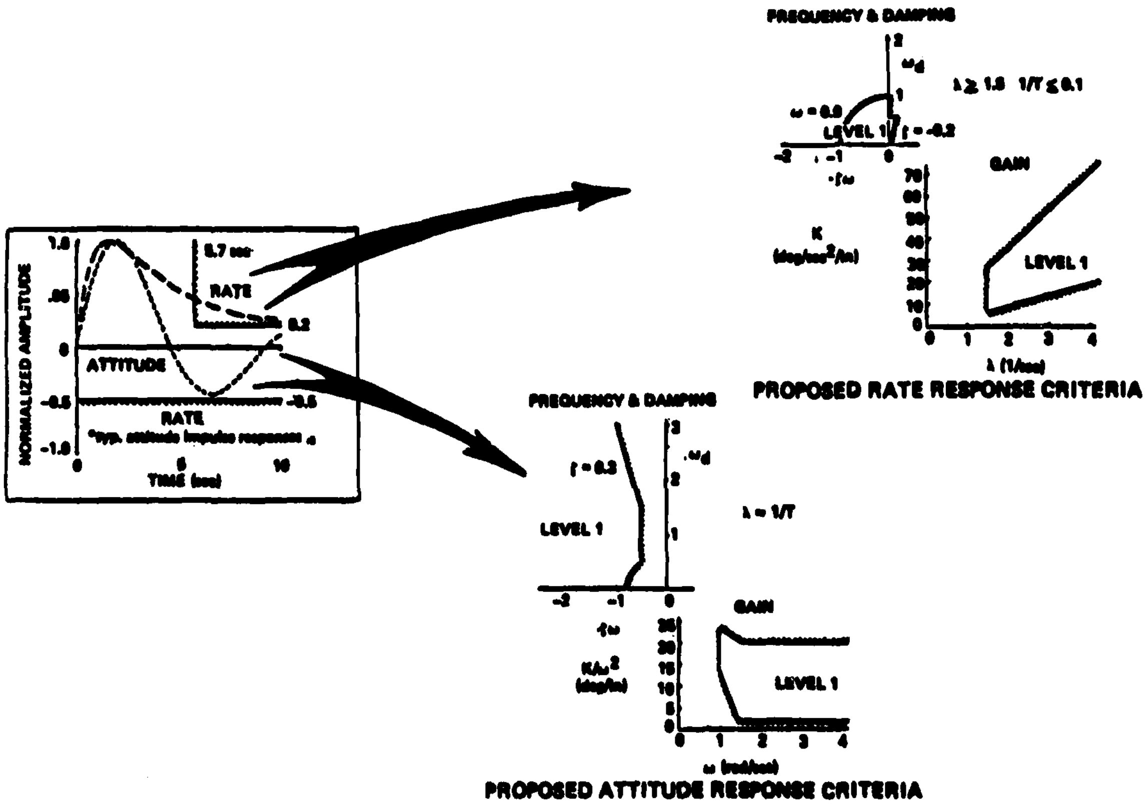


FIGURE 5: PROPOSED ATTITUDE RESPONSE CRITERIA

Once the response type is classified and its LOS equivalent is determined, the following criteria are used to assess dynamic acceptability. For attitude rate systems to be Level 1, the numerator root must be less than or equal to 0.1. The aperiodic denominator root must be greater than or equal to 1.5 and the oscillatory damping and frequency and high frequency gain must satisfy the criteria of Figure 5. Level 1 attitude systems must meet the corresponding frequency, damping and gain requirements of Figure 5. Acceptable levels of time delay have been previously defined.

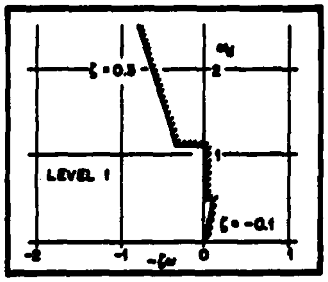
Close comparison of Figure 5 and Paragraph 3.2.2.1 of Table 3 reveals that the combined proposed criteria cover roughly the same region of acceptability as the

TABLE 3: MIL-F-83300 HOVER ATTITUDE STABILITY AND CONTROL REQUIREMENTS

3.2.2 Dynamic response requirements

3.2.2.1 Pitch(roll). The following requirements shall apply to the dynamic responses of the aircraft with the cockpit controls free and with them fixed following an external disturbance or an abrupt pitch (roll) control input in either direction. The requirements apply for responses of any magnitude that might be experienced in operational use. If oscillations are nonlinear with amplitude, the oscillatory requirements shall apply to each cycle of the oscillation.

Level 1: All aperiodic responses shall be stable. Oscillatory modes of frequency greater than 0.5 radians/second shall be stable. Oscillatory modes with frequency less than or equal to 0.5 radians/second may be unstable provided the damping ratio is less unstable than -0.10. Oscillatory modes of frequency greater than 1.1 radians/second shall have a damping ratio of at least 0.3.



3.2.2.2 Directional damping. While hovering at zero airspeed, the yaw mode shall be stable and the time constant shall not exceed the following:

Level 1: 1.0 second

3.2.3 Control characteristics. To insure adequate hover and low-speed control characteristics, the following requirements shall be satisfied starting from flight at constant speed with zero angular rate.

3.2.3.1 Control power. With the wind from the most critical direction relative to the aircraft, control remaining shall by such that simultaneous abrupt application of pitch, roll amd yaw controls in the most critical combination produces at least the attitude changes specified in Table IV within one second from the initiation of control force application.

TABLE IV. Attitude Change in One Second or Less (Degrees)

Level	Pitch	Roll	Yaw
1	+3.0	+4.0	+6.0

3.2.3.2 Response to control input. The ratio of the maximum attitude change, occurring within the first second following an abrupt step displacement of the appropriate cockpit control, to the magnitude of the cockpit control command shall lie within the bounds of Table V. There shall be no objectionable nonlinearities in aircraft response to control deflections and forces.

TABLE V. Response to Control Input in One Second or Less (Degrees per Inch)

	Pitch		Roll		Yaw	
Level	Min	Max	Min	Max	Min	Max
1	3.0	20.0	4.0	20.0	6.0	23.0

3.2.3.3 Maneuvering control margin. When automatic stabilization and control equipment or devices are used to overcome an aperiodic instability of the basic aircraft, both the magnitude of the instability and the installed control power shall be such that at least 50 percent of the nominal control moment can be commanded by the pilot in the critical direction through the use of the cockpit controls.

3.2.4 Control lags. Starting from trimmed hovering or low-speed flight, the angular acceleration response in the commanded direction shall be developed within 0.1 second after the initiation of step displacements of the pitch, roll and yaw cockpit controls. In addition, the initial maximum angular acceleration shall be achieved within 0.3 second after the initiation of the cockpit control command. These requirements apply for input amplitudes of up to 0.5 inches.

existing specification does but with better system definition. Further experimentation is needed to define consistent requirements for Levels 2 and 3.

The third class of hover control augmentation considered is that of translation rate control (TRC). MIL-F-83300 does not cover this type of response but an ES criteria has been proposed to specify satisfactory TRC dynamics. The proposed LOS form is given by equation (4).

$$\frac{\text{translational velocity}}{\text{cockpit control deflection}} = \frac{K\, e^{-\tau s}}{(Ts+1)} \tag{4}$$

The Level 1 boundaries for K and T are given by Figure 6. These boundaries are based on results from an in-flight simulation using the X-22A Variable Stability Research Aircraft (reference 21). Further analysis of these data are required to validate the preliminary boundaries and also to determine the effect of including a transport delay.

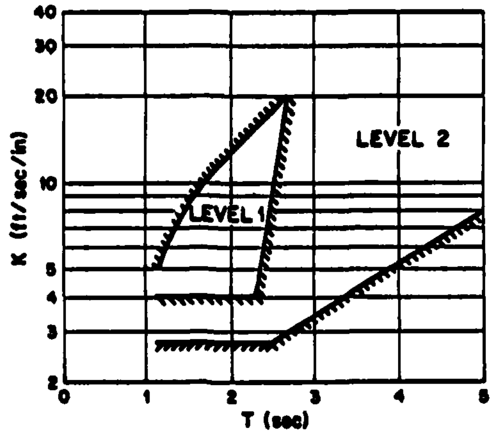


FIGURE 6: PROPOSED TRANSLATIONAL RATE RESPONSE CRITERIA

The hover directional criterion of the specification (Paragraph 3.2.2.2) limits the allowable first order time constant for Level 1 yaw rate response to less than 1 second. This requirement may be applied directly in the ES format if the following LOS form is assumed.

$$\frac{\text{yaw rate}}{\text{cockpit control deflection}} = \frac{K}{(s+1/T)} \tag{5}$$

Acceptable levels of the gain, K, in equation (5) are defined by Figure 7. The requirement of Figure 7 is merely a reformulation of the yaw criterion of Paragraph 3.2.3.2 of MIL-F-83300.

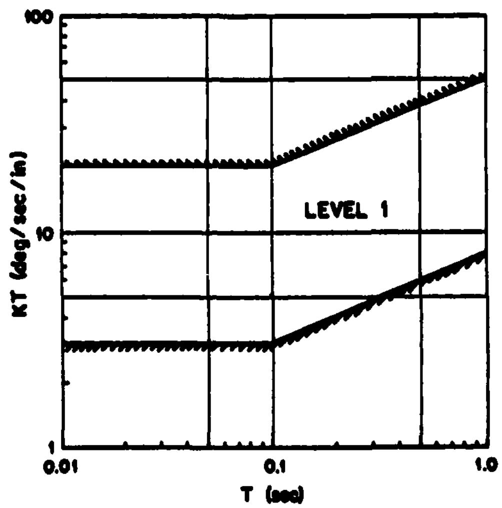


FIGURE 7: YAW RESPONSE CRITERIA

FORWARD FLIGHT/CONVERSION CRITERIA

Revision effort on criteria applicable to the forward flight and transition regimes is nowhere near as far along as that for hover. The hover regime has received the most attention to date because, for the Navy mission, it involves the most severe operating environment and has the most rigorous mission control requirements. Now that hover requirements appear to be better defined, attention is being directed to correcting identified deficiencies in forward flight criteria. In particular, two studies are currently underway which address forward flight equivalent system analysis and criteria.

The first of the studies is being done under contract by McDonnell Aircraft Company. Through analysis and manned simulation, it is addressing aspects of applying the frequency response equivalent system approach to typical VSTOL forward flight system models. Control mode blending characteristics and attitude retention characteristics are also being studied. To whatever extent possible, control power usage during the simulation will be documented. The program is scheduled for completion in July 1982.

The second program is being performed by Systems Technology, Inc., and is concentrating on updating specific criteria. So far, this effort has progressed slowly due to a lack of availability of reliable data. To remedy this situation, a manned simulation effort, specifically dealing with transition flight, is planned for September 1982 at the NASA Ames Research Center. The priority areas planned for investigation include: (1) revision of existing Forward Flight Criteria to allow for application to highly augmented aircraft; (2) investigation of disturbances due to actuation of conversion controls; (3) unique characteristics of final transition to the hover; and (4) blending requirements when transitioning between different augmentation schemes.

This concludes the discussion of fixed-wing criteria development. The remainder of the paper will deal with ongoing and planned efforts specifically addressing rotary-wing criteria.

MIL-H-8501A DEFICIENCIES

In comparison to the VSTOL specification, MIL-F-83300, very little systematic work has been undertaken within the past decade to update the helicopter specification, MIL-H-8501A. As mentioned above the major military helicopter development programs since 1965 have used type specifications designed exclusively for the flying qualities characteristics of a particular vehicle mission and rotor configuration. Although the type specifications were at first basically MIL-H-8501A with slight revisions, recent development of the SH-60B and the AAH have had type specifications very different from MIH-8501A. This is due to the need to address the increased mission requirements of these helicopters. The launch and recovery of the SH-60B from a seaborne platform in up to Sea State 5 conditions is an example of these requirements. Recent work with the HXM type specification highlighted new problem areas, including the need to address any characteristics that may be unique to a tilt-rotor configuration. Through the past decade many papers have been written describing specific areas MIL-H-8501A is deficient (references 8,9,10). The major problem areas described by these papers are discussed in the following paragraphs.

MIL-H-8501A presently addresses helicopter flying qualities in terms of the longitudinal, lateral, directional and vertical axes. There is no systematic delineation between hover/low speed characteristics and forward flight characteristics. In hover a helicopter pilot tends to use longitudinal, lateral, and directional controls independently. For example, in a station keeping task translation along the longitudinal and lateral axes is implemented by the respective cyclic input, while heading angle is controlled by pedal inputs. Forward flight characteristics of a helicopter resemble those of an airplane, thus the pilot needs to use lateral and directional controls in a coupled manner. Also many single rotor helicopters show a coupled pitch-roll dynamic oscillation in hover, whereas in forward flight a dutch roll type of response is characteristic. A breakdown of the helicopter specification into hover/low speed criteria and forward flight criteria (similar to MIL-F-83300) would be a means to address the different axis couplings between hover and forward flight.

A suggestion by Key (reference 8) is that a restructuring of MIL-H-8501A in line with MIL-F-83300 and MIL-F-8785C would allow for a more thorough covering of degraded flying qualities. MIL-H-8501A presently has qualitative criteria for failures of power boosted controls, automatic stabilization systems and engine failures. Table 4 presents one section of a criterion addressing failure of an automatic stabilization system. There is little guidance available on what a sufficient level of control or stability is quantitatively. With the complex augmentation systems being employed on the SH-60B and the CH-53E there is a need to set minimum quantitative levels of degraded flying qualities for partial AFCS failures and single or dual engine failures. The three levels of flying qualities used in the VSTOL and CTOL specifications could be incorporated in MIL-H-8501A to specify quantitative levels of degraded flying qualities for control response, static stability, and dynamic stability in any flight mode.

A third area that could benefit from a restructuring of MIL-H-8501A is in defining criteria that are mission oriented. The helicopter specification currently uses a weight parameter for hover control power considerations that is the result of scaling laws and not meant to represent the variations in control response due to vehicle mission

differences. Both the VSTOL and CTOL specification define four class of vehicles according to overall mission requirements although in MIL-F-83300 the class distinctions are only used for control force limits and roll control effectiveness in forward flight. Table 5 shows a general breakdown of mission as used in MIL-F-83300. Shipboard recovery and nap-of-the-earth (NOE) flight missions could be incorporated into these type of class divisions.

TABLE 4: EXAMPLE OF MIL-H-8501A CRITERIA FOR STABILIZATION SYSTEM FAILURES

3.5.9(d)	Helicopters employing automatic stabilization and control or stability augmentation equipment or both shall possess a sufficient degree of stability and control with all the equipment disengaged to allow continuation of normal level flight and the maneuvering necessary to permit a safe landing under visual flight conditions.

TABLE 5: MIL-F-83300 CLASSIFICATION OF AIRCRAFT

CLASS	DESCRIPTION
I	Small, light aircraft such as - light utility - light observation
II	Medium weight, low-to-medium maneuverability aircraft such as - utility - search and rescue - anti-submarine - assault transport
III	Large, heavy, low-to-medium maneuverability aircraft such as - heavy transport - heavy bomber
IV	High maneuverability aircraft such as - fighter - attack

The probability that future helicopters will have fly-by-wire (FBW) or fly-by-light (FBL) control systems is very high. In conjunction with the FBW/FBL technology it is planned that 4-axis sidestick force controllers will be part of future helicopter control systems. MIL-H-8501A, similar to MIL-F-83300 presently specifies attitude response per inch of control displacement, as well as static stability requirements in terms of control position gradients. The fact that MIL-H-8501A can not give adequate design guidance in the areas associated with sidestick controllers or FBW control systems was highlighted in the development of the HXM type specification. The Army is currently funding a program to fill in the large gap of necessary handling qualities data to develop criteria addressing level 1 flying qualities with FBW/FBL sidestick controllers, as well as degraded flying qualities due to SCAS failures.

The Navy has begun an independent program assessing the basic flying qualities criteria in MIL-H-8501A against the VSTOL specifications (MIL-F-83300 and AGARD 577) and representative present and future rotary wing aircraft. The significant results from the assessment of hover control power criteria and dynamic response criteria are presented in the following sections.

HOVER CONTROL POWER

Helicopter control power requirements are usually determined by hover control mission requirements. As described above, MIL-H-8501A uses a weight parameter to specify attitude response within one second or less. In an extensive review of MIL-H-8501A, Walton and Ashkenas (reference 6) suggest that the MIL-H-8501A weight dependency is too simplified to give adequate guidance for various vehicle missions. On the other hand, the two VSTOL specifications specify a constant limit of attitude response. A comparison of the VSTOL specification boundaries and the MIL-H-8501A requirement for roll attitude per inch of lateral control displacement as a function of the vehicle gross weight is shown in figure 8. The lower boundaries of all three

specifications are substantiated by the level 2 rating given to the XV-15 with augmentation off. There are two other major points to be raised from figure 8. First the CH-53D AFCS on response has been described as quite adequate for the assault mission, yet the vehicle does not satisfy the VSTOL boundary. This then substantiates the need for some type of weight dependency as used by MIL-H-8501A. It is questionable whether or not pilots will accept a lower response for extremely large vehicles like the HLH (GW=130000 lb). A vehicle in this weight category would only need to attain a bank angle of 2.1 degrees within one second for a one inch lateral stick displacement to satisfy the MIL-H-8501A requirement. The second point from figure 8 is the large difference in roll response between the similar weight SH-60B and CH-46A (ten degrees per inch versus four degrees per inch). The CH-46A has been described as having very adequate response characteristics for its assault and vertical replenishment missions. The SH-60B has been qualitatively described as having just adequate response characteristics for a turbulent, high sea state condition, characteristic of the LAMPS mission. Yet the SH-60B shows a response well above the visual flight rules (VFR) or instrument flight rules (IFR) MIL-H-8501A boundary. The difference between these two vehicles then raises the point of having attitude response criteria a function of mission and weight. In particular the small landing platforms and dynamic atmosphere conditions Navy helicopters will be expected to launch and recover from are an example of a mission that may not be adequately designed for by the still wind, out-of-ground effect control power criteria in MIL-H-8501A.

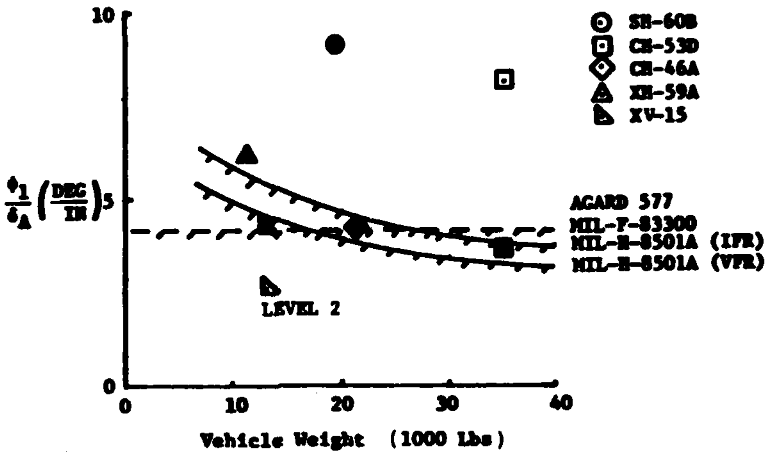


FIGURE 8: HOVER ROLL RESPONSE COMPARISONS

To insure that the helicopter response is not initially too sensitive MIL-H-8501A also has minimum angular rate criteria for the longitudinal, lateral, and directional axes. Using these damping boundaries with the above attitude response criteria, rate damping versus sensitivity boundaries can be developed. Figure 9 shows the ABC and tilt-rotor compared to the MIL-H-8501A requirements for the yaw axis. The interesting point here is that neither aircraft satisfied the requirement yet the ABC has been described in a recent Navy flight test program as having "crisp, predictable" yaw control and that the "high yaw rates (in excess of 45 degrees per second) that resulted from one inch pedal step inputs were well-damped and easily arrested, allowing large, rapid heading changes." The XV-15 in comparison was described as sluggish and not adequate. The ABC develops yaw control through differential collective of the two rotor systems while the tilt-rotor develops yaw control via fore and aft nacelle tilting. The results shown in figure 9 show an apparent anomaly between MIL-H-8501A and the different rotor configuration of the ABC and tilt-rotor. Figure 10 shows the pitch response characteristics of the SH-60B, CH-53D and the XV-15. Similar to the directional axis MIL-H-8501A adequately predicts the single rotor vehicle ratings (the SH-60B and the CH-53D) but again the tilt-rotor shows a discrepancy.

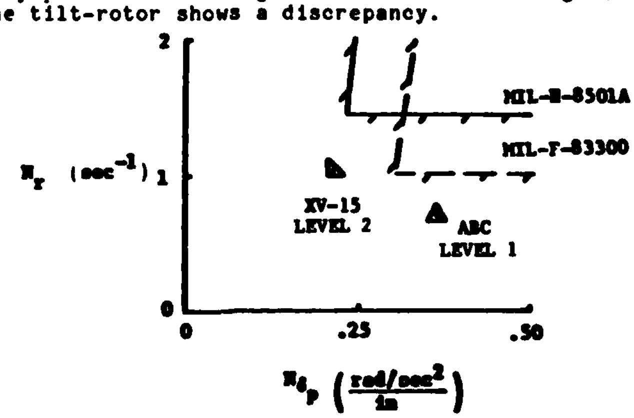


FIGURE 9: YAW RATE VS. SENSITIVITY COMPARISONS

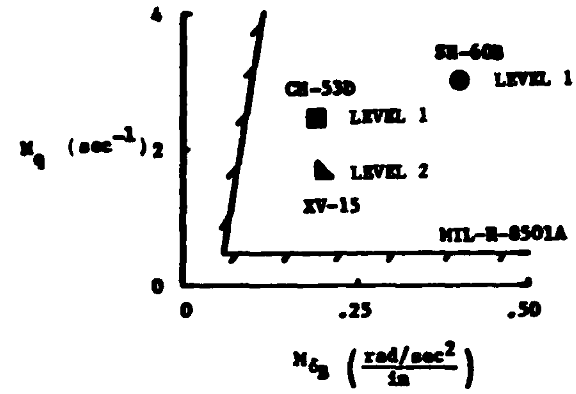


FIGURE 10: PITCH RATE VS. SENSITIVITY COMPARISONS

Overall it was found that the MIL-H-8501A attitude response and angular rate damping criteria gave minimal design guidance in comparison to the vehicles analyzed. Further analysis and data are needed to determine the effect of vehicle mission and varied rotor configurations.

DYNAMIC STABILITY

Following a disturbance (control or atmospheric) to a helicopter in hover the above rate damping criteria should ensure an initial satisfactory response. After this initial response the aircraft may still have an unacceptable dynamic response. In a precision hover task it is mandatory that the pilot be able to easily correct for unwanted oscillatory responses. Uncommanded pitch or roll responses can cause tracking or station keeping errors, plus any short period dynamic responses must be well-damped so as not to impede precise control of the helicopter.

Satisfactory boundaries for dynamic stability characteristics are defined by each of the specifications reviewed through the use of second-order response parameters. The general trend is similar for all the specifications such that short period oscillations require a damped response while for longer periods, neutral stability to slight instability is acceptable. Figure 11 shows a plot of nondimensional damping ratio versus damped natural period with a comparison of the three specifications for pitch or roll hover dynamic responses. Note that only MIL-H-8501A has a separate boundary for VFR conditions. As mentioned above, it is assumed within MIL-F-83300 that "IFR capability is inherent in all military aircraft operational missions." For the limited data available very few conclusions can be drawn about the adequacy of the specifications boundaries. Of the three aircraft shown only the SH-60B shows a "conventional" phugoid mode. Reference 7 presents the point that for modern helicopters

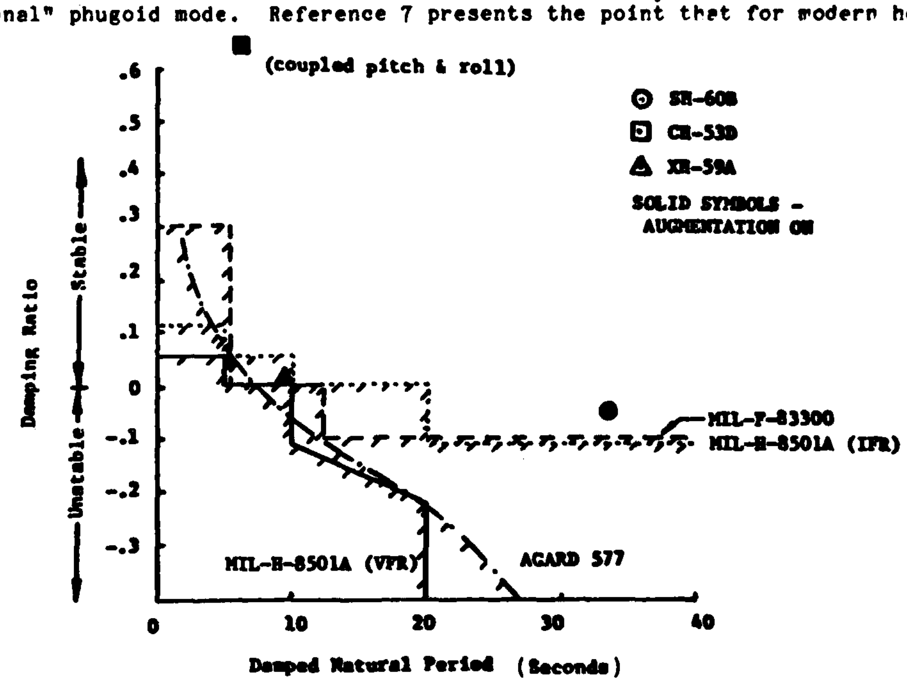


FIGURE 11: HOVER LONGITUDINAL DYNAMIC STABILITY REQUIREMENTS

the figure 11 MIL-F-83300 boundary is generally undemanding. This is questionable considering the SH-60B response that Navy pilots described as adequate for the LAMPS mission. Both the CH-53D and the XH-59A have also been qualitatively described as having level 1 characteristics. In particular the CH-53D has essentially dead-beat dynamic responses in hover. From the data analyzed it appears that MIL-H-8501A gives adequate guidance for hover dynamic responses.

Just as in hovering conditions, it is necessary that a helicopter have satisfactory dynamic response characteristics in forward flight. For example, in contour flying or mine sweeping missions, a slowly divergent phugoid response with a gradual altitude loss would be objectionable. MIL-H-8501A specifies VFR and IFR dynamic response criteria for the longitudinal axis (the same as the above hover requirements), while only stipulating IFR criteria for the lateral-directional axes.

Looking first at the longitudinal criteria, figure 12 shows a comparison between the VSTOL and helicopter specification boundaries. The helicopter specification is by far the most lenient in specifying stability requirements, in particular for long period responses (≥20 seconds) under VFR conditions. In contrast, the VSTOL specifications do not allow divergent long period dynamic responses. With augmentation on, the three vehicles shown on figure 12 easily satisfied all the specifications. Each aircraft has also been given level 1 ratings, in particular the SH-60B is described as having excellent phugoid damping It should be noted that both VSTOL specifications have additional requirements for short period oscillations such that the damping ratio must be at least 0.3. AGARD 577 defines a short period response such that the damped period is less than 3 to 6 seconds. MIL-F-83300 specifies short period requirements according to figure 13. Note that the frequency boundary is a function of the vehicle n/α ratio. The CH-53D was the only vehicle analyzed that showed a short period type response, and it compared favorably with the figure 13 boundaries (e.g. $\zeta > 0.3$). For the vehicles compared against MIL-H-8501A, the specification gives lenient but adequate guidance for normal flight conditions.

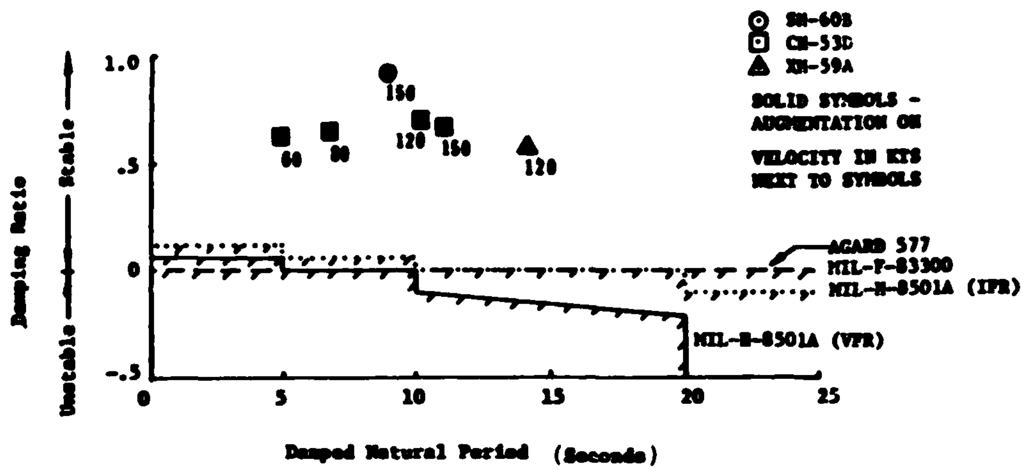


FIGURE 12: FORWARD FLIGHT LONGITUDINAL DYNAMIC STABILITY REQUIREMENTS

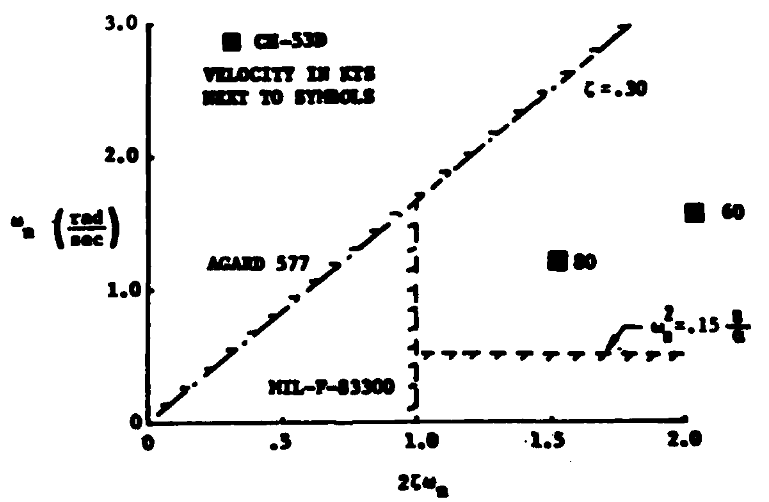


FIGURE 13: VSTOL SPECIFICATION SHORT PERIOD REQUIREMENTS

The lateral-directional dynamic stability requirements as specified by the VSTOL and helicopter specifications are shown on figure 14. The same general trend is followed by each criterion. Note that MIL-H-8501A has no requirement for VFR lateral-directional dynamic stability. The cluster of open symbols shows a common damped dutch roll response for the single rotor helicopters (SH-60B, CH-53D, SH-3A) analyzed. This type of yaw-roll coupled dynamic response has been given unsatisfactory ratings for single rotor helicopters. Thus there should at least be a baseline criteria limiting allowable divergent responses for VFR conditions. For augmentation on the responses are all well-damped over a wide range of frequencies. An interesting comparison between varied rotor configurations is shown on figure 14 as the ABC has a dutch roll response that falls right on the MIL-F-83300 level 1 boundary. Pilots described the ABC as having very satisfactory lateral-directional forward flight characteristics that were very similar to a fixed wing aircraft. A Sikorsky report (reference 22) on the ABC compared this response to MIL-F-8785, the fixed wing flying qualitites specification. The ABC again appears as an anomaly in comparison to the helicopter specification boundary. For the vehicles analyzed MIL-H-8501A gives adequate guidance for IFR lateral-directional dynamic responses but has no guidance for VFR conditions.

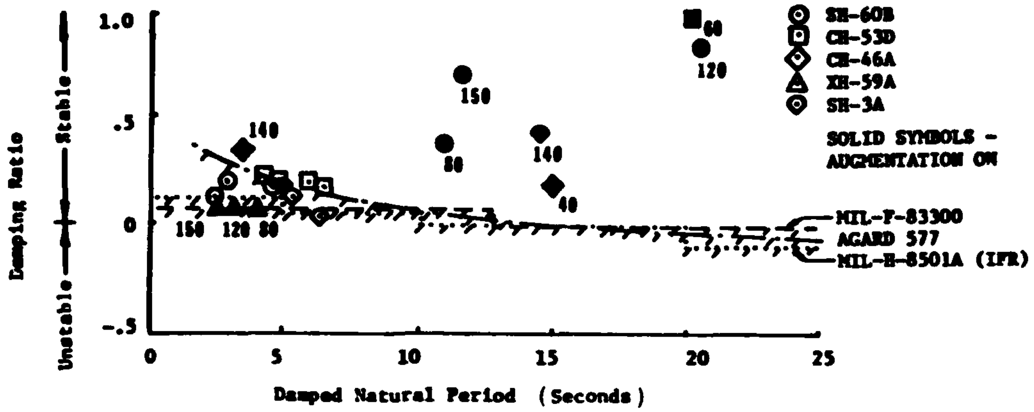


FIGURE 14: FORWARD FLIGHT LATERAL-DIRECTIONAL DYNAMIC STABILITY REQUIREMENTS

CONCLUSIONS

FIXED-WING

Since the acceptance of MIL-F-83300, sufficient insight has been gained to warrant the revision of some sections and the addition of others to the specification. This revision process must be an interactive one including inputs from all elements of the VSTOL community - both government and industry. This paper has presented the status of recent developments in the establishment of revised criteria. Specifically:

Available visual cue information level must be adequately defined and included within the specification of minimum control augmentation and display levels.

Equivalent, low order system definition appears to be a viable approach to the specification of levels of hover stability and response of highly augmented VSTOL configurations.

Unique specification of response parameters for attitude rate, attitude and translational rate augmentation is required.

Continuing work is planned to validate proposed criteria revisions through in-flight simulation. Other areas of the VSTOL flight envelope (i.e., transition and conversion) are also being examined.

ROTARY-WING

Although the need to update MIL-H-8501A has been known for many years, very little systematic work has been directed towards developing modern criteria. A step towards this goal is the future Army-Navy program designed to develop an updated rotary-wing handling qualities specification. This paper has presented the major deficiences in MIL-H-8501A as cited by many previous papers as well as the significant results of a preliminary Navy assessment of MIL-H-8501A. In particular:

MIL-H-8501A does not give adequate guidance to address the

differences in handling qualities characteristics between hovering and forward flight conditions.

MIL-H-8501A has very limited guidance for degraded flying qualities, especially towards defining minimum characteristics for AFCS failures.

The hover control power criteria (attitude response and rate damping criteria) inadequately address varied mission characteristics or rotor configuration differences.

Dynamic response criteria are in general adequate but very lenient, in particular for VFR mission requirements where no guidance is given for lateral-directional responses.

Analyses in the areas of height control response, aerodynamic and gyroscopic cross-coupling characteristics, and autorotation criteria are underway.

REFERENCES

1. Anonymous, "Flying Qualities of Piloted VSTOL Aircraft," MIL-F-83300, Dec 1970.

2. Scheuren, MAJ W. J. and Dunn, J. L., "Flying Qualities and Performance Trials of the Model AV-8A Airplane," Naval Air Test Center Report FT-27R-71, 28 Apr 1971.

3. Lacey, T. R., "MIL-F-83300; View from an Aircraft Designer," Proceedings of the Navy/NASA VSTOL Flying Qualities Conference, pp 21-52, Apr 1977.

4. Anderson, S. B., "A Comparison of the VSTOL Handling Qualities of the VAK-191B with the Requirements of AGARD Report 577 and MIL-F-83300," NASA TP-1494, Jul 1979.

5. Anonymous, "Helicopter Flying and Ground Handling Qualities; General Requirements for," MIL-H-8501A, 7 Sep 1961.

6. Walton, R. P. and Ashkenas, I. L., "Analytical Review of Military Helicopter Flying Qualities," Systems Technology, Inc., Technical Report No 143-1, Aug 1967.

7. Green, D. L. and Richards, R. B. Jr., "Review of MIL-F-83300 'Flying Qualities of Piloted VSTOL Aircraft' to Assess the Applicability to Helicopters," Pacer Systems, Inc., Technical Report No PWR-054-72, Mar 1972.

8. Key, D. L., "A Critique of Handling Qualities Specifications for US Military Helicopters," AIAA Paper 80-1592, Aug 1980.

9. Dooley, L. W., "Handling Qualities Considerations for NOE Flight," American Helicopter Society Journal, Vol 22, No 4, Oct 1977.

10. Pitt,D. M. and Heacock, F. E., "Advanced Scout Helicopter Flying Qualities Requirements, How Realistic Are They?" AHS Report 79-28, May 1979.

11. Anonymous, "VSTOL Handling Qualities Criteria, Part I - Criteria and Discussion Dec 1970; Part II - Documentation," NATO AGARD Report 577, Jun 1973.

12. Hoh,R. H. and Ashkenas, I. L., "Identification of Priority Deficiencies in MIL-F-83300 and Possible Improvements Required for Navy Sea-Based Operations," STI Working Paper No 1116-1, Feb 1978.

13. Anonymous, "Flying Qualities of Piloted Aircraft," MIL-F-8785C, Nov 1980.

14. Anonymous, "NAVTOLAND Navy Vertical Takeoff and Landing Program," Naval Air Systems Command, NAVTOLAND Project Office, Sep 1976.

15. Anonymous, "VSTOL Displays for Approach and Landing," AGARD R-594, Jul 1972.

16. Lebacqz, J. V. and Aiken, E. W., "A Flight Investigation of Control, Display and Guidance Requirements for Decelerating Descending VTOL Instrument Transitions Using the X-22A Variable Stability Aircraft," Calspan Corp., Report AK-5336-F-1, Sep 1975.

17. Lebacqz, J. V., Radford, R. C. and Beilman, J. L., "An Experimental Investigation of Control/Display Requirements for a Jet Lift VTOL Aircraft in the Terminal Area," Calspan Corp., NADC 76099-60, Jul 1978.

18. Hoh, R. H. and Ashkenas, I. L., "Development of VTOL Flying Qualities Criteria for Low Speed and Hover," Systems Technology, Inc., NADC 77052-30, Sep 1979.

19. Hodgkinson, J. and LaManna, W. J., "Equivalent Systems Approach to Handling Qualities Analysis and Design Problems of Augmented Aircraft," AIAA Paper No 77-1122, Aug 1977.

20. Carpenter, C. G. and Hodgkinson, J., "VSTOL Equivalent Systems Analysis," McDonnell Aircraft Co., NADC 79141-60, May 1980.

21. Radford, R., "An Experimental Investigation of VTOL Flying Qualities Requirements for Shipboard Landings," Calspan Corp., NADC 77318-60, Jun 1981.

22. Ruddell, A. J., et al, "Advancing Blade Concept (ABC) Technology Demonstrator," USAAVRADCOM-TR-81-D-5, Apr 1981.

EQUIVALENT SYSTEMS CRITERIA FOR HANDLING QUALITIES OF MILITARY AIRCRAFT

J. Hodgkinson
MCDONNELL AIRCRAFT COMPANY
MCDONNELL DOUGLAS CORPORATION
P.O. BOX 516
ST. LOUIS, MO 63166

<u>Introduction</u> - Low order equivalent systems appear viable for mapping high order augmented systems into a lower-dimensional form suitable for specifying flying qualities. Degrees of allowable mismatch between high and low order systems are defined in tentative new criteria. Alternative specification methods, such as the Neal-Smith method and the bandwidth method, are fundamentally similar to equivalent systems. Because the alternative methods involve mapping, they too exhibit mismatch.

<u>Simulation and Specification for Aircraft</u> - In the past, the short term pitch rate dynamics of aircraft were readily represented by this linear second order response to stick force.

$$\frac{\dot{\theta}}{F_S} = \frac{K(s + 1/T_{\theta 2})}{s^2 + 2\zeta\omega s + \omega^2} \equiv \frac{(1/T_{\theta_2})}{[\zeta;\omega]} \quad (1)$$

Texts such as Reference 1 define all of the terms in this function, using aerodynamic characteristics, aircraft speed, inertia, etc. This expression ignores some other higher order terms which have to be assessed separately - for example, structural modes, stick dynamics, and the interaction of other aerodynamic modes of motion such as the phugoid. The effects of these terms were small, as were the effects of linearization; or if they were significant, they could be considered separately. Thus this transfer function defined a well-accepted mathematical "simulation space" in which the response was defined.

All of the parameters in this response appeared (explicitly or implicitly) in the forerunners of the flying qualities Military Specification MIL-F-8785C (Reference 2). Therefore, this set of parameters was entirely suitable for defining flying qualities regions. The four parameters K, T_{θ_2}, ω_{SP}, and ζ_{SP} defined a well-accepted "specification space".

For past aircraft, the dimension of the simulation space and the specification space was the same. In emerging designs the flying qualities were determined by interpolation between known research data points in the space. Incidentally, it requires about four pages in MIL-F-8785 to specify flying qualities in this four-dimensional space.

<u>Simulation and Specification for Modern Aircraft</u> - Figure 1 illustrates a pitch rate transfer function which emerged during development of the F-18 flight control system. The simplifying assumptions are similar to those used in Equation (1). This is a digital system and some digital effects, such as aliasing filters, are included, but the effects of sample and hold and computational delay are not. Given that the designers will make allowance for these elsewhere, Figure 1 was the accepted simulation model of that particular system. The simulation space for this example had 89 dimensions. Following our experience with past aircraft, we might estimate that to define flying qualities regions in the 89-dimensional specification space would take 89 pages in MIL-F-8785. Clearly another approach was needed.

$$\frac{\dot{\theta}}{F_S} = \frac{[-0.5;127]\,[-0.3;83]\,[-0.07;61]\,[1;0.9]\,[1;1]\,[1;1.2]\,[1;2.3]\,[1;2.8]\,(4)\,[0.1;53]\,(13)\,(14)}{(0.5)\,(0.8)\,(0.9)\,[1;1.1]\,(2)\,[1;2.4]\,[0.7;4.1]\,(3.4)\,(3.7)\,(6.7)\,[0.2;53]\,[1;13]\,[0.4;33]\,[0.7;20]} \cdots$$

$$\times\ \frac{[0.4;34]\,[0.34;46]\,[0.55;39]\,[1;22]\,[0.7;37]\,[0.9;34]\,[0.9;34]\,(35)}{[0.3;42]\,[0.9;18]\,[0.6;27]\,[0.3;66]\,[0.4;85]\,[0.4;105]\,[0.9;47]\,[0.3;140]\,[0.4;180]} \cdots$$

$$\times\ \frac{1}{[0.7;102]\,(72)\,[0.5;228)\,(112)\,[0.6;279]\,[0.8;329]\,(259)\,[0.9;378]\,[1;410]} \cdots$$

X *TERMS DUE TO DIGITAL SYSTEM*

GP83-0117-15

FIG. 1 MODERN HIGH ORDER RESPONSE

Finding alternative methods of specifying the flying qualities of these high order systems is a challenge. Analytical tools have emerged so that high order systems can be specified in fewer dimensions. Examples for short-term pitch dynamics are: the equivalent system approach, which approximates high order responses with low order responses; the Neal-Smith method, which uses a mathematical model of the pilot; and the bandwidth method, which takes simple frequency response measurements. If the gain parameter K is specified separately, equation (1) has a three-dimensional specification space. The other two methods are only two-dimensional. Human pilots, of course, process these systems to produce a one-dimensional specification of flying qualities, namely, the Cooper-Harper rating, as shown in Figure 2.

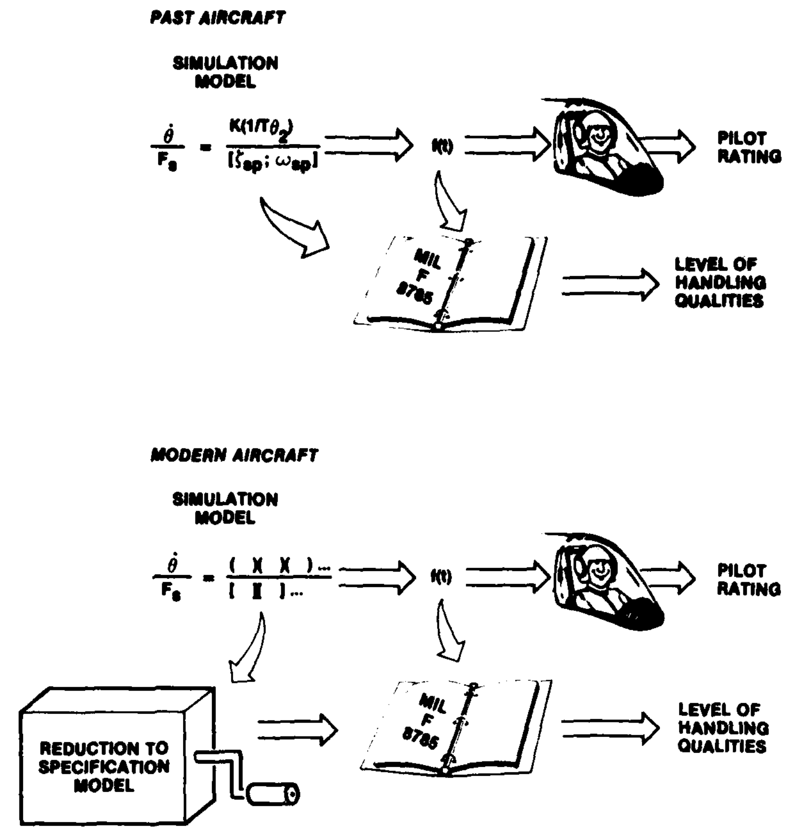


FIGURE 2
FLYING QUALITIES SPECIFICATION COMPLIANCE FOR PAST AND MODERN AIRCRAFT

Equivalent Systems - As mentioned, one approach to specifying high order responses such as that of Figure (1) is offered by the use of equivalent systems. MIL-F-8785C requires manufacturers to match the high order frequency responses with lower order classical forms (e.g., equation 1). A computer program is used for matching.

Most of our experience has been with longitudinal equivalent systems, though matched equivalent systems are required for lateral-directional dynamics also. Matching disallows the often misleading procedure of extracting a subset of the high order roots for evaluation.

Equivalent Time Delay - Early equivalent systems studies by DiFranco, Neal and Smith, and MCAIR (References 3, 4, and 5) determined quickly that the specification space of classical systems defined by Equation (1) was insufficient for high order systems. For one thing, the phase lags of high frequency modes were not accounted for. As mentioned in Reference 1, "for low-pass inputs the major gross effect of the highest frequency modes, assuming that they are well beyond the crossover frequency, is an initial time delay."

As Figure 3 points out, an equivalent time delay approximates the high frequency terms quite well. Surprisingly, in the DiFranco and Neal and Smith investigations, these delays consistently degraded pilot ratings down. Therefore, time delay must be used in any viable equivalent system, adding one extra dimension to the specification space in MIL-F-8785. Equivalent delays of .1, .2 and .25 seconds are the upper Level 1, 2, and 3 limits. More recently we have defined phase delay, τ_p (Figure 4). This enables the engineer to extract the time delay effect from a phase frequency response by hand. Usually, τ_p is numerically very similar to equivalent delay.

Some idea of the importance of delay effects is gained from regression analysis of Neal and Smith's 61 configurations. The simple equation

$$\text{Pilot rating} = 3.7 + 24.8\ \tau_p$$

has a correlation coefficient of .70. The multiplier of 24.8 on τ_p shows that it would take only 40 milliseconds of equivalent delay to degrade a configuration by one Cooper-Harper point.

The later data of Smith (Reference 6) were analyzed also. The equation was

$$\text{Pilot rating} = 2.5 + 25.3\ \tau_p$$

The correlation coefficient was 81%.

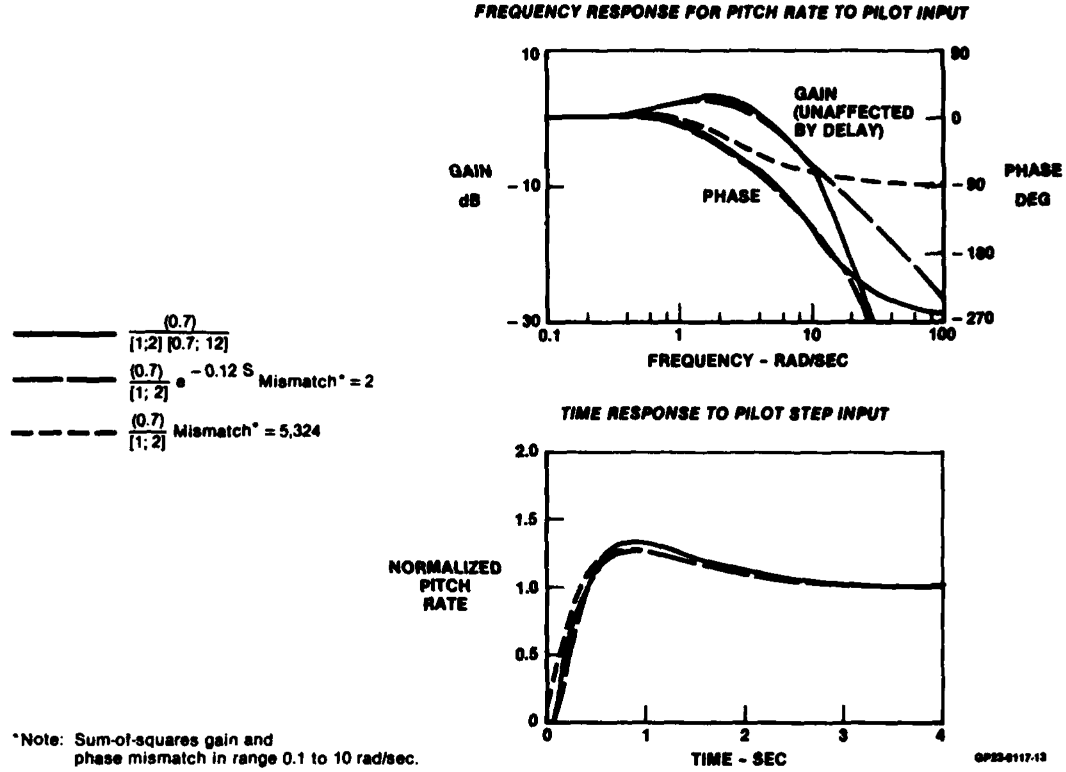
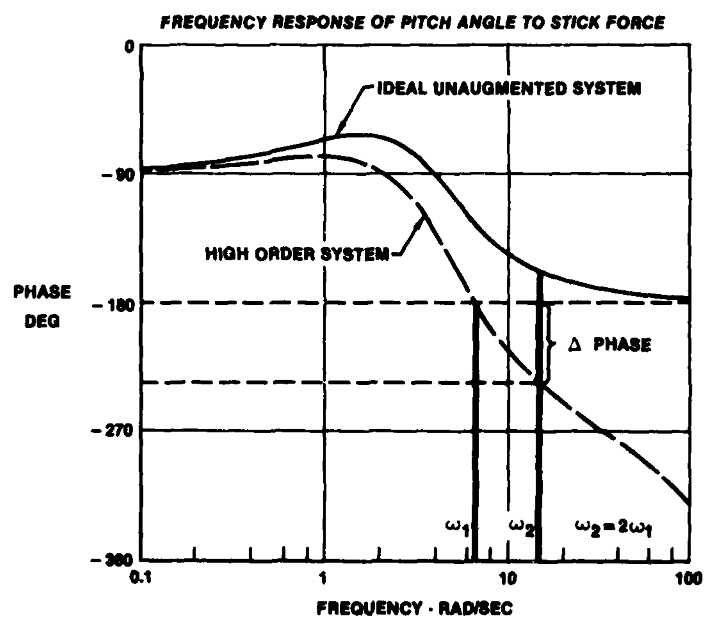


FIGURE 3
COMPARISON OF EQUIVALENT SYSTEMS WITH AND WITHOUT TIME DELAY



$$\tau_p = \text{PHASE DELAY (SEC)} = \frac{\Delta\,\text{PHASE (RAD)}}{\omega_2} = \frac{\Delta\,\text{PHASE (DEG)}}{57.3\,\omega_2},\ \omega_2\ \text{IN RAD/SEC}$$

GP23-0117-2

FIGURE 4
DEFINITION OF PHASE DELAY, τ_p

These correlations show that the Calspan experiments of References 4 and 6 successfully defined the added specification dimension needed for augmentation systems. These experiments are the basis of the equivalent systems work which eventually led to the formal requirements of *MIL-F-8785C*.

In the V/STOL experiment of References 7 and 8, we varied both time delays and gain in the command path. Figure 5 illustrates that without time delay, a high gain produced an excellent rating but even small delays led to pilot-induced oscillations. Similar results (to be published by Calspan shortly) have been obtained in flight using the CTOL NT-33.

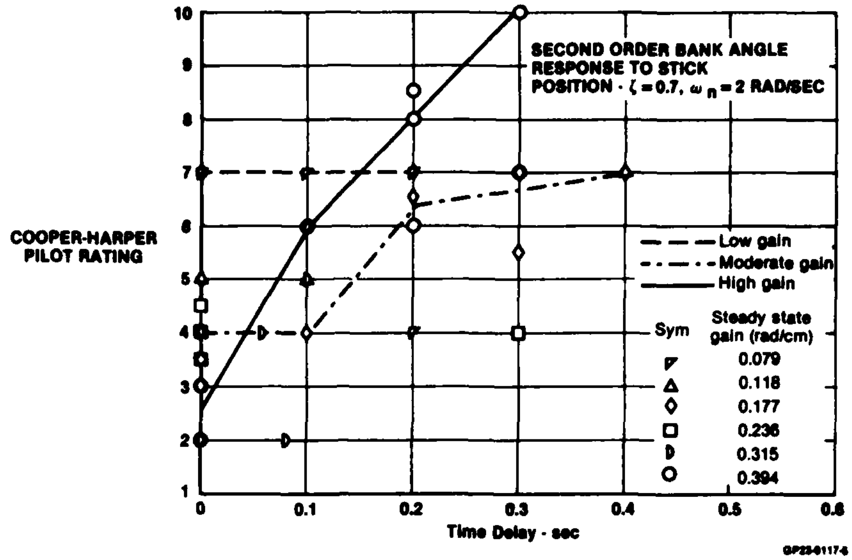


FIGURE 5
INTERACTION OF STEADY STATE GAIN AND TIME DELAY

The pilot rating degradation due to delays (as discussed in Reference 9) has not been predicted consistently in ground-based simulation. Exaggerating the simulated command gain is one possible way of forcing the pilot into the task and producing more reliable rating degradations in ground-based simulators.

Equivalent delays are used to approximate the effects of many different flight control components. These include stick prefilters, feel systems, actuators, and structural filters. Perhaps surprisingly, digital systems need not be large contributors. Sample and hold times, computational delays, aliasing prefilters and postfilters add to the "overhead" of delay which must be accounted for. But these have often produced less equivalent delay than analog components.

Equivalent System Mismatch Envelopes - Because equivalent low order systems approximate high order systems, there is a need to define what differences are allowable in order to evaluate the precision of match. Valuable exploratory data were gathered in a joint USAF/USN/MCAIR simulation in 1978, using the Calspan variable stability NT-33 (References 10 and 11). We designed high order and low order equivalent systems with mismatches of various magnitudes in various frequency ranges. We expected large mismatches to be directly correlated with pilot rating differences. Instead, we could see little correlation. High order systems had the same ratings as their low order equivalents, even if the mismatch was large.

When we performed a later study for V/STOL aircraft, we took these data into account. On that basis it seemed unlikely that our ratings would show the desired differences if we simply minimized the frequency response difference between the high and low order systems. So, this time we simulated local mismatches in various frequency ranges by adding higher order terms to low order systems. By doing this progressively - for example, adding more and more high frequency lag - we were able to note the point where pilot ratings became significantly different on the two systems.

We summarized these fixed-base V/STOL simulation results (References 7 and 8) by drawing an envelope around the frequency responses of the added terms. The result (Figure 6) suggested a reason for the pilot's insensitivity to mismatch in the earlier NT-33 study. Even though the mismatches were far larger than those usually generated by a computer program, they were still within the envelopes and therefore remained largely unnoticed.

Calspan, in the CTOL, in-flight simulation studies of Neal and Smith (Reference 4) and Smith (Reference 6) also evaluated high order effects by adding them to low order systems. Although in their experimental design they did not have mismatch in mind, we were able to use their results to generate envelopes (Reference 12). The envelopes (Figure 7) appear in the proposed flying qualities Military Standard, Reference 13. Again, the envelopes are quite large. When we reanalyzed the NT-33 data of References 10 and 11, those mismatches which were not noticed generally fell within the envelopes.

The envelopes emphasize the importance of a good match in a central frequency region (about 2 to 4 radians per second). We therefore modified our computer program so that the match process can be weighted more heavily in the central region. In practice, the weighting changed the equivalent system parameters very little.

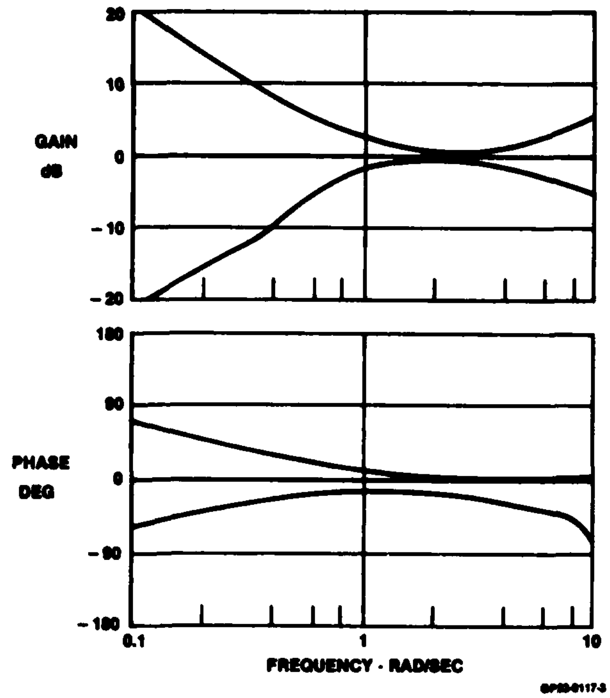


FIGURE 6
ENVELOPES OF MAXIMUM AUGMENTATION
ADDED WITHOUT AFFECTING RATING

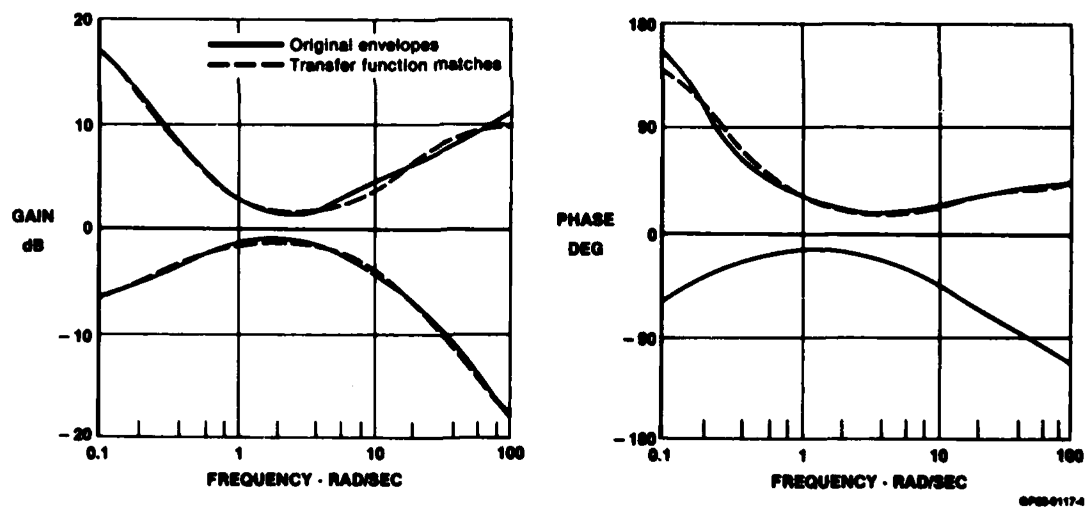


FIGURE 7
CTOL ENVELOPES AND TRANSFER FUNCTION MATCHES

In the experiments mentioned so far, the high order terms were added to the low order systems one at a time. However, it is recognized that an equivalent may have more than one distinct region of mismatch. We would hope that any such system, even if the mismatches fell within the envelopes, would be questioned by the procuring activity.

<u>The Mismatch Problem</u>

It is frequently said that a fundamental drawback of equivalent systems is the need to define an acceptable level of mismatch.

Other criteria have emerged, however, such as the Neal-Smith and bandwidth method, which simply use the response of a system, regardless of its order, and do not involve a matching process. Therefore it is frequently said that the question of mismatch does not arise, which is a fundamental advantage.

However this statement turns out to be quite untrue. This can be seen by viewing any specification method heuristically as a mapping process, as follows.

Specification space is the image space, and the simulation space is the inverse image space. For every system in the image space, there is an infinitely large inverse image set in the inverse image space. "Mismatch" is the difference between any member of the inverse image set and a "minimum order system" which maps one-to-one onto the image space.

As an example, a specific equivalent system, with given specification parameters, could result from a number of different higher order systems. Mismatch defines the difference between any of these higher order systems and the equivalent system. For example, sum-of-squares frequency response mismatch (a scalar with a value of about 10 or less for a visually acceptable match) has been defined in Reference 5. The minimum order system for the equivalent system method is the equivalent system itself.

The challenge is to find minimum order systems which, conceptually, span the specification space of other specification methods.

Neal-Smith Method - As shown in Figures 8 and 9 the Neal-Smith method (Reference 4) uses pilot lead or lag compensation as a measure of system, and closed-loop resonance as a measure of vulnerability to pilot-induced oscillation. A pilot model is used which maintains a fixed pilot-in-the-loop bandwidth.

CLOSED-LOOP RESPONSE MODEL IN NEAL-SMITH CRITERION

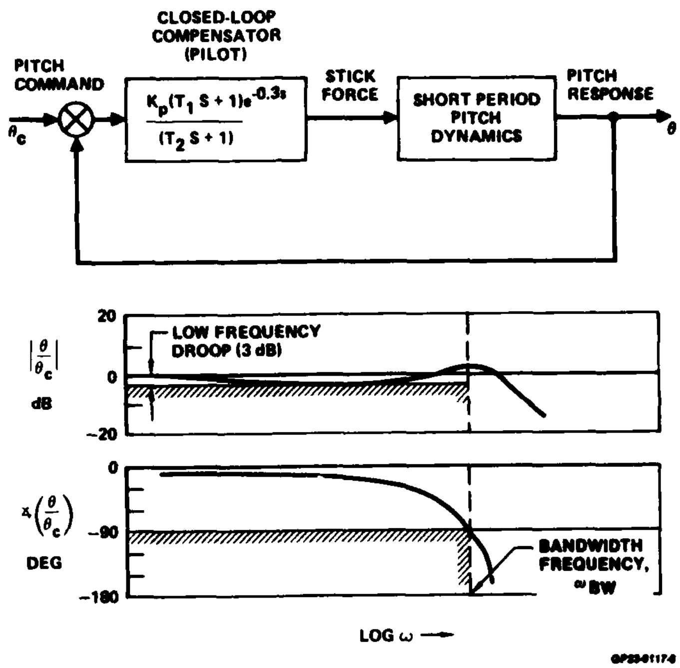


FIGURE 8
NEAL-SMITH DEFINITIONS

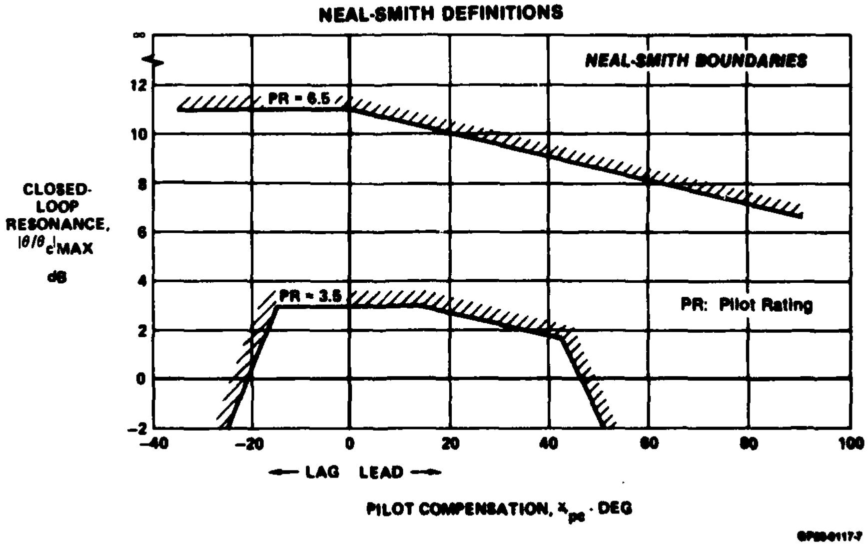


FIGURE 9
NEAL-SMITH CRITERION

Figure 10 compares pilot lead with equivalent short period frequency. The two parameters are closely related, as would be hoped.

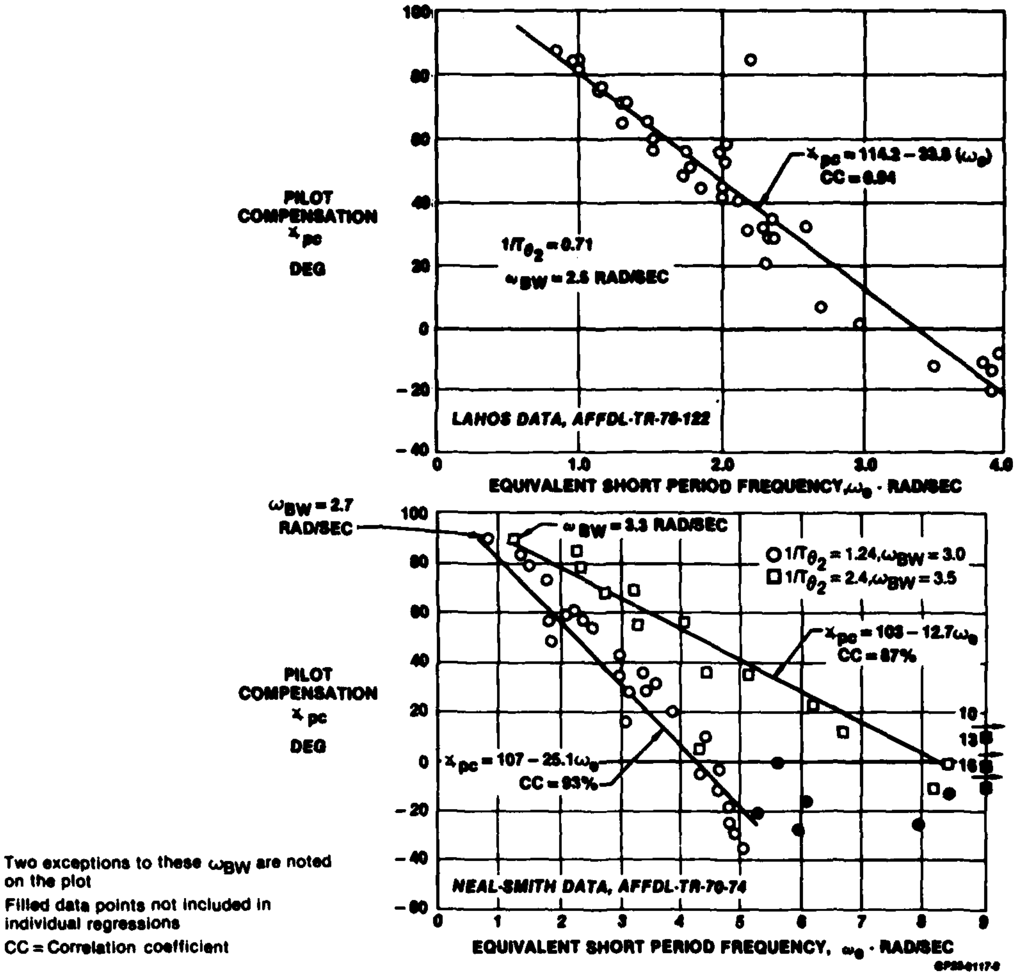


FIGURE 10
PILOT COMPENSATION vs EQUIVALENT SHORT PERIOD FREQUENCY

More complex, but equally valid, correlations were found between Neal-Smith and other equivalent system parameters: in particular, equivalent delay. Apparently the Neal-Smith and equivalent systems methods complement each other while producing essentially similar information (Reference 14).

Finding a minimum order system for the Neal-Smith method is not simple. We examined various transfer functions and allowed a time delay as part of the form. Preliminary results suggest that different order systems are required for different regions of the Neal-Smith specification space. However, a large region of the space is mapped by

$$\frac{\theta}{F_S} = \frac{K_\theta e^{-\tau s}}{(Ts + 1)}$$

and this is sufficient to demonstrate an example of mismatch for the Neal-Smith method.

Figure 11 shows three high order configurations with essentially the same parameters in the Neal-Smith specification space. Two are high order systems, and one is the minimum order system.

According to the Neal-Smith specification parameters, the mismatches shown in Figure 11 (1108 for configuration 2-2, and 1116 for 5-4) should be insignificant to the pilot. Equivalent systems separately determined for the two high order configurations in Reference 15 obtained mismatches of 24 and 2 respectively. Because of their higher dimensionality, equivalent systems produced less mismatch than the Neal-Smith method. The Neal-Smith method probably produced similar specification parameters for these three different systems because they are dynamically similar in the 2 to 3 radians/second region.

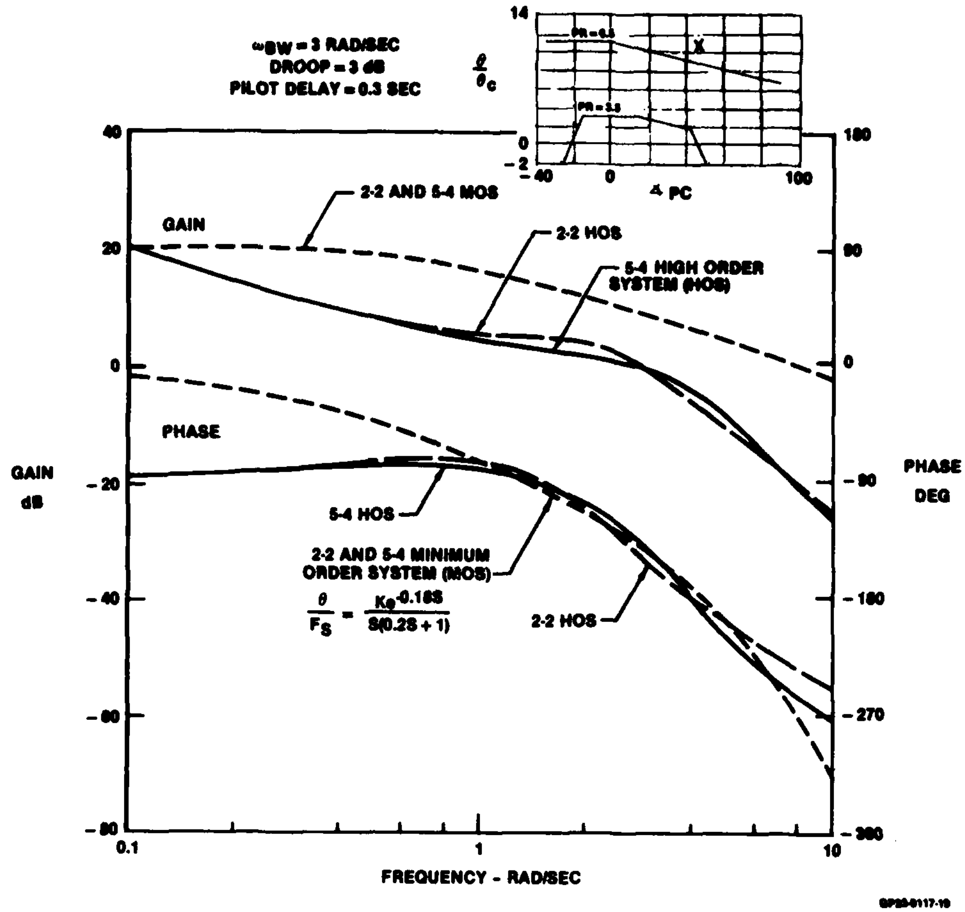


FIGURE 11
FREQUENCY RESPONSE COMPARISON FOR TWO HIGH ORDER SYSTEMS AND ONE MINIMUM ORDER SYSTEM, ALL WITH SIMILAR NEAL-SMITH CRITERION PARAMETERS

<ins>The Bandwidth Method</ins> - Figure 12 defines bandwidth as proposed for the flying qualities Military Standard and Handbook. It reverses the Neal-Smith philosophy by defining the bandwidth for gain-alone pilot compensation. It is used with phase delay, defined in Figure 4, to specify flying qualities (Figure 13). Figure 14 shows the high correlation of bandwidth frequency with equivalent system frequency.

For the bandwidth method, the minimum order system is:

$$\frac{\theta}{F_S} = \frac{K_\theta e^{-\tau S}}{S(TS + 1)}$$

Figure 15 shows two different high order systems and their minimum order system. Again, these all have the same specification parameters according to the bandwidth method. The phase responses are very similar in the 45° phase margin region (about 2 to 3 radians/second). The mismatches are 519 and 84 for configurations 2-3 and 4-11 respectively. According to Reference 15 equivalent system mismatches were far lower -36 and 0.3. Again, the higher-dimensioned method produces the lowest mismatch.

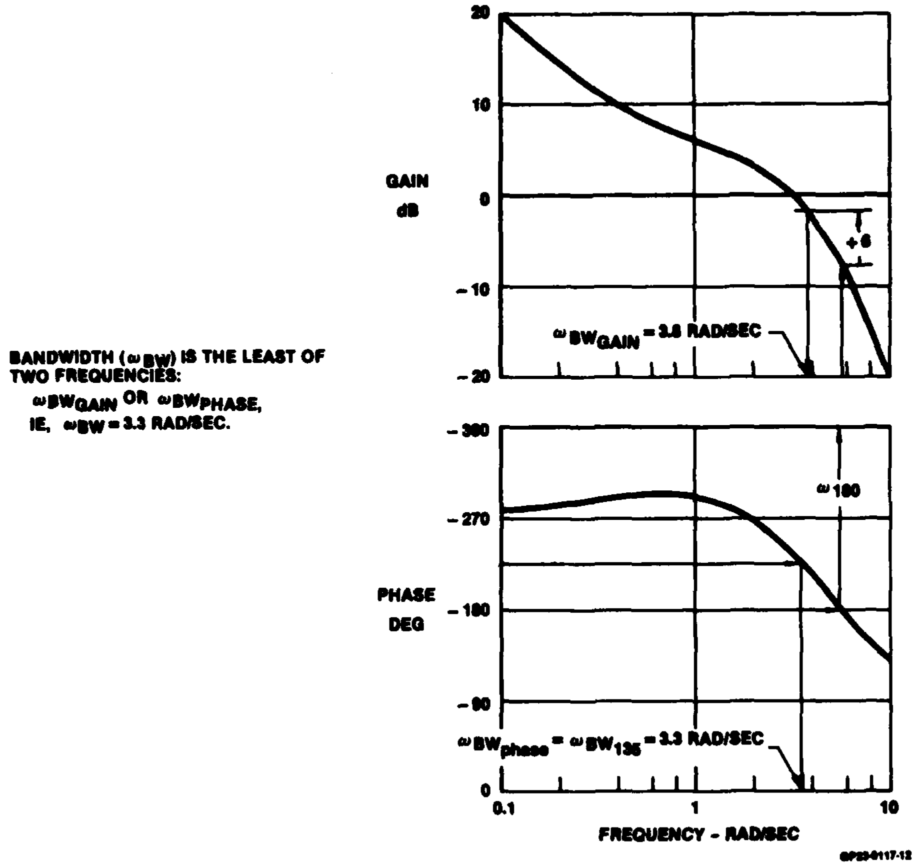


FIGURE 12.
DEFINITION OF BANDWIDTH FOR PITCH ANGLE
RESPONSE TO STICK FORCE

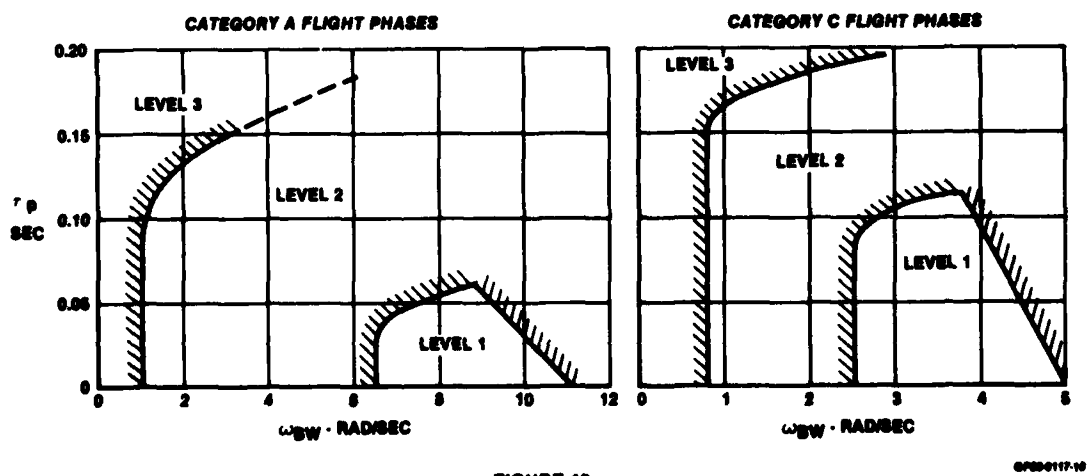


FIGURE 13
BANDWIDTH REQUIREMENTS FROM PROPOSED
MIL HANDBOOK

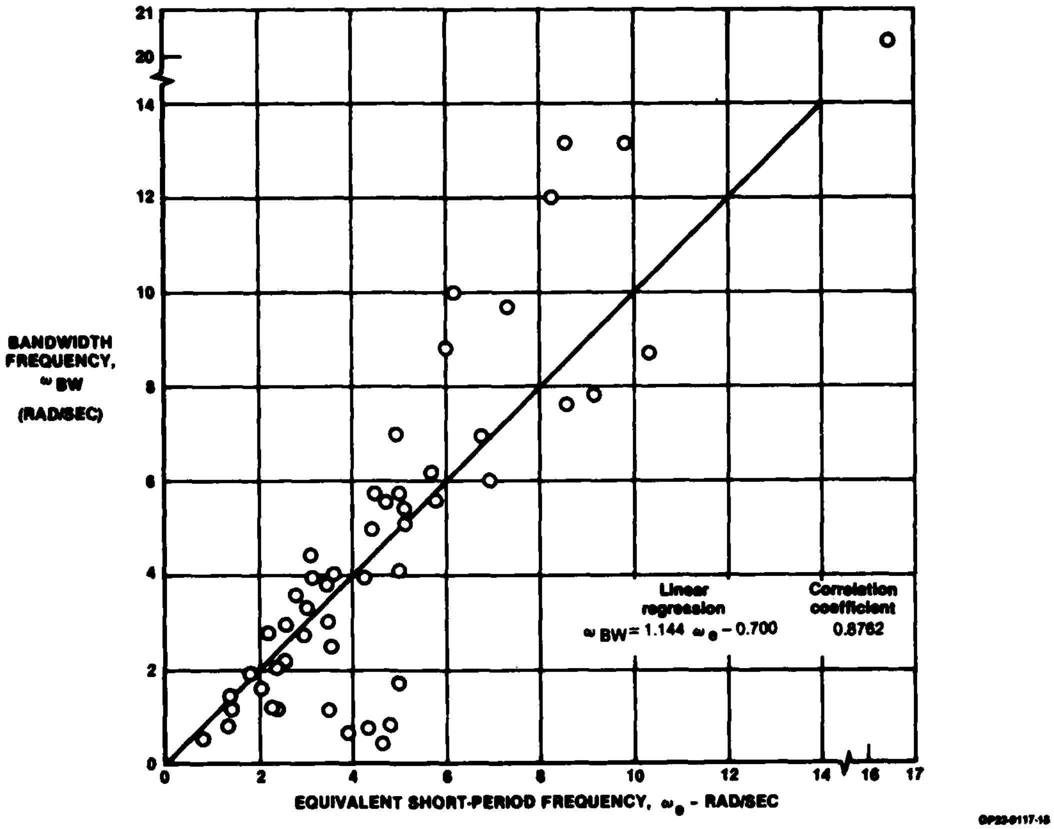


FIGURE 14
COMPARISON OF EQUIVALENT SYSTEM FREQUENCY AND BANDWIDTH

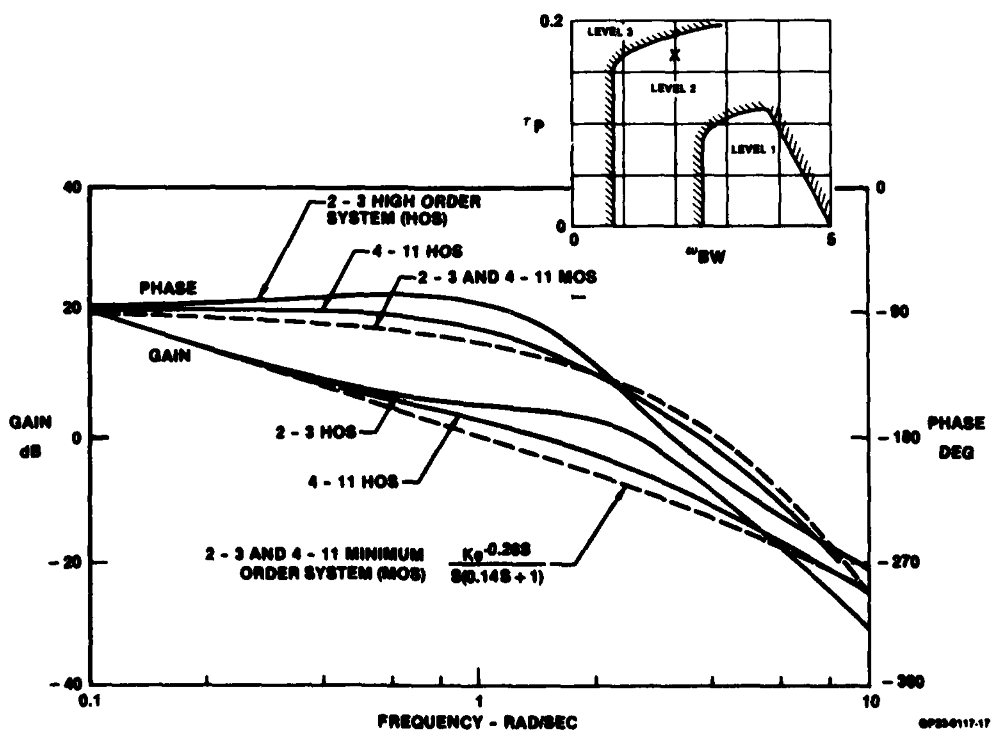


FIGURE 15
FREQUENCY RESPONSE COMPARISON FOR TWO HIGH ORDER SYSTEMS AND ONE MINIMUM ORDER SYSTEM, ALL WITH SIMILAR BANDWIDTH AND τ_p VALUES

Conclusions - Low order equivalent systems appear to be suitable for specifying the flying qualities of high order augmented systems, as proposed in Reference 16. Even though augmentation systems increase the dimensions of the specification space, an equivalent time delay term can be used to approximate these effects.

However, other methods also can be used to produce similar information, though their underlying assumptions may appear different.

Any method for specifying high order systems involves a reduction in dimension. This means that some high order effects are ignored. These effects are quantified as "mismatch" in determining equivalent systems. Mismatches can also be quantified for the other methods, such as the Neal-Smith and bandwidth method, using concepts of mapping and a 'minimum order system'. Therefore, mismatch should not be viewed as peculiar to the equivalent system method.

Predictably, the more a specification method reduces the dimension, (i.e., the lower the order of its minimum order system) the larger the mismatch can be. Thus the two-dimensional Neal-Smith and bandwidth methods typically produce more mismatch (i.e., ignore more high order effects) than the four-dimensional equivalent system method.

All methods for high order systems are in a real sense equivalent system methods.

Acknowledgements

The contributions of J. A. Koegler and J. R. Wood of MCAIR are acknowledged.

REFERENCES

1. McRuer, Duane; Ashkenas, Irving; and Graham, Dunstan, "Aircraft Dynamics and Automatic Control," Princeton University Press, 1973.

2. Moorhouse, D. J., and Woodcock, R. J., "Background Information and User Guide for MIL-F-8785C, Military Specification-Flying Qualities of Piloted Airplanes," AFWAL-TR-81-3109, June 1981.

3. DiFranco, D. A., "In-Flight Investigation of the Effects of Higher-Order Control System Dynamics on Longitudinal Handling Qualities," AFFDL-TR-68-90, August 1968.

4. Neal, T. P., and Smith, R. E., "An In-Flight Investigation to Develop Control System Design Criteria for Fighter Airplanes," AFFDL-TR-70-74, Vols. I and II.

5. Hodgkinson, J., LaManna, W. J., and Heyde, J. L., "Handling Qualities of Aircraft with Stability and Control Augmentation Systems - A Fundamental Approach," J. RAES, February 1976.

6. Smith, R. E., "Effects of Control System Dynamics on Fighter Approach and Landing Longitudinal Flying Qualities," AFFDL-TR-78-122, March 1978.

7. Carpenter, C. G., and Hodgkinson, J., "V/STOL Equivalent Systems Analysis," NADC-79141-60, May 1980.

8. Carpenter, C. G., Clark, J. W. Jr., Hodgkinson, J., "Quantification of V/STOL Equivalent System Characteristics Through Analysis and Ground-Base Simulation," AIAA Atmospheric Flight Mechanics Conference, Danvers, Mass., August 1980.

9. Hodgkinson, J., and Snyder, R. C., "Flight Evaluation of Augmented Fighter Aircraft," AIAA Atmospheric Flight Mechanics Conference, Danvers, Mass., August 1980.

10. Smith, R. E., Hodgkinson, J., and Snyder, R. C., "Equivalent System Verification and Evaluation of Augmentation Effects on Fighter Approach and Landing Flying Qualities," AFWAL-TR-81-3116, September 1981.

11. Hodgkinson, J., and Johnston, K. A., "Initial Results of an Inflight Simulation of Augmented Dynamics in Fighter Approach and Landing," AIAA Guidance and Control Conference, Boulder, Colorado, August 1979.

12. Wood, J. R., and Hodgkinson, J., "Definition of Acceptable Levels of Mismatch for Equivalent Systems of Augmented CTOL Aircraft," MDC A6792, 19 December 1980.

13. Hoh, R. H., et al, "Proposed MIL Handbook - Handling Qualities of Piloted Airplanes," AFWAL TR (Preliminary) December 1981.

14. Hodgkinson, J., "Comparison of Two Flying Qualities Design Criteria for Advanced Flight Control Systems," in Crombie, R. B., and Moorhouse, D. J., "Flying Qualities Design Criteria," Proceedings of AFFDL Flying Qualities Symposium held at WPAFB October 1979.

15. Johnston, K. A., and Hodgkinson, J., "Flying Qualities Analysis of an In-Flight Simulation of High Order Control System Effects on Fighter Aircraft Approach and Landing," MDC Report A5596, 22 December 1978.

16. A'Harrah, R. C., Hodgkinson, J., and LaManna, W. J., "Are Today's Specifications Appropriate for Tomorrow's Airplanes?", AGARD Flight Mechanics Panel Symposium on Stability and Control, Ottawa, Canada, September 1978.

Piloted Handling Qualities Design Criteria for High Order Flight Control Systems

by

John C. Gibson
British Aerospace
Public Limited Company
Aircraft Group, Warton Division
Warton Aerodrome, Preston, England.

SUMMARY

In the period broadly under review, the 1970's, several aircraft have been brought into service which utilise full authority fly-by-wire (FBW) control systems, firstly with analogue techniques and more recently with digital computing. While many benefits have been obtained, it is true to say that such systems have not always reached their full potential to provide handling qualities superior to much simpler aircraft of the past. In particular, sluggish response and pilot induced oscillations (PIO) in both pitch and roll axes typify the experience with a high proportion of FBW aircraft.

To avoid such problems it is necessary to maintain a clear understanding of the basic response characteristics of "low order" aircraft, that is aircraft with insignificant control system dynamics. The relationships between these are discussed and satisfactory ranges derived from in-flight experiments are presented. It is shown how "high order" aircraft responses can be directly and simply related to "low order" requirements as expressed in MIL-F-8785. Several criteria are presented which permit adjustment of handling qualities for some specific tasks, including one assuring satisfactory control at touchdown.

While this paper is almost exclusively confined to the pitch axis, similar methods can be applied to the roll axis. These methods are in routine use at British Aerospace and are proving highly effective in providing excellent handling qualities.

1. INTRODUCTION

The most significant change affecting handling qualities in the Seventies has been the widespread adoption of FBW control systems with full control authority, replacing the small authority stability augmentation systems which were mainly effective in small perturbation manoeuvres. FBW has given the flight control system (FCS) designer the power to modify handling to an extent never before possible, more especially with newer digital FCS where complex gain scheduling and filtering, cross-feed paths, multiple feedbacks etc. become feasible. It should have been possible to ensure outstandingly good handling on all these aircraft, but sometimes experience has failed to match expectation.

In contrast, there have always been examples of low order aircraft through aeronautical history rated as "pilots' aircraft" because of their excellent handling qualities. As performance increased so that powered controls became necessary, problems including PIO's arose caused by valve forces, actuator lags, spring-and-bobweight pitch feel, and so on. Solutions to all these problems were found during the 1950's. Especially noteworthy were the efforts by STI and others to determine pilot-aircraft closed loop behaviour and PIO causes by pilot modelling. By the early 60's this work was essentially complete in that the knowledge available then is still largely sufficient to avoid most of today's handling problems.

It remained true that "pilots' aircraft" were those departing least from traditional low order characteristics. A classic example is the Lighting, still in service with high performance even by today's standards, whose prototype flew in 1954, which explored its complete flight envelope initially and safely without stability augmentation, including very high speeds "on the deck" despite being designed as a high altitude supersonic interceptor. Close attention to the quality of its control circuits, actuator performance, use of hydraulic pitch Q-feel and a simple 3-axis damper system were combined with its aerodynamic qualities into an aircraft always greatly liked by its pilots for its handling qualities.

With today's FCS technology it has proved easy to eliminate the uncommanded nuisance responses of past aircraft, for example the pitch trim changes with flaps, reheat, airbrakes and transonic effects, dutch roll and sideslip excitation in coarse manoeuvres, and so on. It is impressive to a pilot to be able to open the throttle at low speed and simply await the arrival of maximum speed without touching the stick - but it is not essential to his mission task. It <u>is</u> essential that when he does touch the stick the response is predictable in attitude and flight path, and it is here that it has proved equally easy to introduce undesirable characteristics. Invariably these turn out to be unlike traditional ones, and the question to be answered is "What are the desirable traditional characteristics?".

MIL-F-8785B specified such characteristics and provided a large body of back-up information and guidance. It was the intention of 8785C and will be the intention of the new revision to require aircraft with any FCS to have essentially the same handling qualities. While the spirit of such a complex specification can sometimes be beaten by an absolutely strict adherence to the letter, such an intention is unquestionably the right one in general terms. Because it is mostly defined by low order parameters, for example the pitch short period frequency and damping, difficulties have often arisen in its application to a high order FCS where such parameters are either not identifiable or are merely part of several which dominate the response.

To resolve this difficulty, two approaches have been developed at Warton. The first is a time domain one, in which pilot ratings and especially comment data from in-flight simulation have been correlated to the five basic step response time history features seen by the pilot, which are normal acceleration, flight path angle, pitch attitude, pitch rate and pitch acceleration. The broad outline of 8785 can be divided into subsets of desirable ranges for combat manoeuvring, precision tracking, flight refuelling and landing approach.

The second approach is a frequency domain one in which optimum frequency response boundaries are derived for precision attitude tasks accounting for optimum pilot model characteristics. This was done using the above in-flight comment data and the pilot-vehicle systems analysis methods of 20 years ago.

The result is a range of time response boundaries, frequency response boundaries, and parameter limits which are applicable to aircraft with any order of FCS and which are direct and easy to apply.

2. FUNDAMENTAL TIME DOMAIN RESPONSES

A large part of any flight is conducted in an open loop precognitive manner. The pilot is able to apply discrete, step-like inputs which more or less exactly produce the desired aircraft response. Some closed loop tracking many also be required if continuous disturbances prevent the desired steady flight path from being achieved, either "pursuit" where it is possible to distinguish the effects of his control inputs from the external effects or "compensatory" when this is not possible. Tight closed loop operation is of course necessary in target tracking, and sometimes in instrument flight if the aircraft is very susceptible to turbulence upsets. While closed loop operation depends upon the frequency domain response characteristics, successful precognitive control requires the time domain response to lie within acceptable limits. Failure to satisfy this may well result in additional closed loop control being forced upon the pilot as he endeavours to compensate by overdriving or smoothing unsatisfactory time responses.

Fig. 1 shows the five basic pitch step time responses which collectively determine the predictability and nature of the pilot's task. In particular, the attitude response has an important characteristic labelled either drop back or overshoot. These terms describe the behaviour as the control input is removed when the attitude approaches the values required. If the attitude stops and returns to some previous value, it is called drop back, while if it continues on to some increased value it is called overshoot. These characteristics can of course be measured equally well at the time of the initial control input but they are not of the same direct significance to the pilot because the response is moving in the desired direction. Their magnitude is determined by the ratio of the areas in the pitch rate response labelled A and B.

Fig. 2 shows the relationships of normal acceleration, flight path angle and attitude in more detail. The short period frequency and damping determine the time response of normal acceleration. Integration of this gives the flight path angle time response. The attitude response leads flight path by the time constant $T_{\theta 2}$ in the attitude transfer function (Table 1) and is greater by the increment in angle of attack required to produce the normal acceleration. These elementary relationships are fixed by the aerodynamics of the aircraft and are independent of the FCS. They therefore provide a means of assessing the handling qualities of any aircraft provided that satisfactory characteristic limits can be defined.

Extrapolation of flight path angle response back to the time axis defines what is in effect an equivalent flight path time delay, positive to the right of the initial time. This delay is independent of input and response amplitude for a linear system. Similarly the attitude response yields a constant time parameter equal to the attitude drop back divided by pitch rate, positive to the left. Attitude overshoot results in a negative time parameter to the right of the initial time. In either case the sum of the attitude and flight path time parameters equals $T_{\theta 2}$. Examination of the two series of in-flight simulation by Calspan, Refs. 1 and 2, has resulted in indications of satisfactory task-related values for these time parameters.

Before these are discussed it is necessary to consider how low order short period frequency and damping requirements can be related to aircraft responses where they cannot be identified meaningfully, as is often the case with a high order FCS. Fig. 3 shows in simplified boundary form the unit time response of normal acceleration at the 8785 low and high damping limits. The low damping boundary includes only the first overshoot, the subsequent decaying oscillation having little effect on the mean flight path angle. When these boundaries are integrated into the unit flightpath angle responses it can be seen that for any given frequency, the flight path time delay can vary by a factor of nearly six depending on the damping ratio.

The 8785 Level 1 boundaries for frequency have been redrawn to give instead the length of the time response associated with the permitted frequency, Fig. 4. Here the slowest response is given by the low frequency high damping limits and the fastest by the high frequency low damping limits. Any aircraft response which lies within these boundaries satisfies the intent of the low order 8785 requirements and does not need the equivalent of frequency and damping to be identified.

Some additional provisos should be noted. Any oscillations after the first overshoot must have an amplitude ratio per half cycle of less than 0.3 to satisfy the minimum damping. The first overshoot should not breach the upper right hand boundary which represents the low frequency low damping limit. The Control Anticipation Parameter (CAP) implicit in 8785 frequency limits need to be explicitly stated, that is that the initial pitch acceleration divided by steady g should lie between 0.28 and 3.6 rad/sec^2/g. Finally, a response with pronounced "hang on", that is for example one which follows closely the left hand and upper boundary limits (very unlike a second order response, but not impossible with a high order FCS) would introduce a significant flight path drop back with a negative value of flight path delay. The effect of this is indicated in Fig. 5. It is probably desirable to specify positive delay values to prevent this.

3. DESIRABLE TIME DOMAIN RESPONSE

8785 does not explicitly place bounds on either the time or frequency response of pitch attitude. Given that most aircraft have had sufficiently similar response dynamics, then empirical frequency and damping limits have tended to result in acceptable attitude behaviour. The latter's importance in determining response predictability, closed loop tracking performance, PIO tendencies and so on has been recognised and investigated for over two decades. The advent of full authority FCS has made the absence of attitude criteria unacceptable for several reasons:-

- Additional high order response modes can result in poor or dangerous attitude behaviour.

- Attitude behaviour can be easily modified and so it is necessary to understand its acceptable limits.

- The historic link between between flight path and attitude, that is $T\theta 2$, can be broken by the use of additional direct lift control.

Calspan in-flight simulations of high order FCS pitch handling using their NT33 aircraft have provided an invaluable source of information about acceptable attitude and flight path characteristics. In Ref. 1 the tasks of air combat manoeuvring, air to air tracking and flight refuelling and in Ref. 2 the approach and landing tasks were assessed for wide ranges of basic frequency and damping modified by stick prefilters to simulate high order effects.

Step responses were calculated for all these configurations and their features compared with pilot rating and comment data. Some quite clear results were obtained which can be summarised as follows:-

- Negative attitude drop back (i.e. overshoot) was associated with sluggish, unpredictable response both in flight path control and in tracking.

- Attitude drop back from 0 to about 0.25 seconds was excellent for fine tracking and was associated with comments typified by "the nose follows the stick".

- Increasing attitude drop back led to abrupt response and bobbling, from "slight tendency" to "continuous oscillations", in tracking tasks. Sometimes this was called PIO but it did not cause concern for safety.

- Attitude drop back had little effect within the range tested upon gross manoeuvring without target, landing approach or flight refuelling, provided it was not negative.

- CAP up to 3.6 rad/sec^2/g was satisfactory for gross manoeuvring without a target, but was unsatisfactory above 2 rad/sec^2/g for the landing approach, above 1 rad/sec^2/g for fine tracking, and below 0.28 rad/sec^2/g for any task.

- The pitch rate overshoot ratio seems to qualify the drop back behaviour, with a value greater than 3.0 resulting in unacceptable drop back as small as 0.25 seconds. The trends are indicated in Fig. 6.

- Small values of flight path time delay were associated with excellent flight refuelling control, but were not essential for good gross manoeuvring and did not on their own ensure predictable behaviour.

- Overshoots in normal acceleration did not cause unpredictable behaviour unless associated with low frequency, breaching the upper right hand time response boundary of Fig. 4.

The above factors point to a number of design criteria and trend indicators which have been put to use in control law developments with excellent results. No attempt has been made to define the equivalent of level 1, 2 or 3 limits and it may be that the most appropriate location would be in the 8785 Handbook, for example as background guidance. However, one result stands out clearly as a candidate for a new requirement. This is to specify that attitude drop back be zero or positive, as negative values were always rated sluggish and unsatisfactory in these experiments. This leads also to the result that some pitch rate overshoot is always necessary for optimum handling, with the function of minimising drop back for tracking inputs or of rapid generation of the angle of attack increment required for crisp flight path response in gross manoeuvres, landing flare, etc. It has no significance otherwise for the pilot unless it becomes too large.

It is possible to construct a low order boundary set connecting all three pitch response parameters, as shown in Fig. 7. This is a modified version of a proposal in the revised 8785 Handbook now being reviewed (Ref. 3), and fits the supporting data presented there equally well. From Fig. 2 it can be seen that for zero attitude drop back the equivalent flight path time delay must be $T\theta 2$ seconds. For any short period frequency this delay is fixed by the damping ratio, as seen in Fig. 3. The bottom righthand boundary of Fig. 7 is the resulting zero drop back limit. The bottom lefthand boundary is a PIO criterion based on pilot-airframe closed loop attitude tracking taken from Ref. 4 and which is discussed later in the section on frequency response characteristics. If valid it must represent a level 3 limit and therefore some margin is required from it for level 1, though no PIO data are available to fix the amount, and the boundary drawn here is based on the CAT.A data points from Ref. 3.

The short period frequency required to achieve zero attitude drop back is expressed in Fig. 8 as a proportion of the minimum Level 1 Cat.A frequency permitted by 8785, for three damping ratios and several altitudes over a typical range of wing lift slope and wing loading combinations. It can be seen that heavy damping permits the widest achievement of this characteristic and that this is increasingly difficult as altitude and wing loading are increased and lift slope decreased.

The conclusion to be drawn is that for low order aircraft with elementary pitch damper augmentation, a low manoeuvre margin should be aimed for, with its inherently high natural damping, if precision pitch handling is required. This is completely consistent with the excellent Lightning low altitude high speed (LAHS) pitch handling characteristics where in fact a frequency lower than the 8785 minimum is satisfactory, together with only 2.0 lb/g stick forces. The much larger TSR2 prototype also had a low manoevure margin with good damping in the LAHS region and was taken on only its 20th flight to 550 knots at 250 feet over hilly terrain without any stability augmentation. It was rated as having control and response well matched to this task.

Apart from its use as an indicator of desirable manoeuvre margin trends for one specific task requirement the Fig. 8 criterion need not be followed very strictly. If the 8785 minimum frequency is adhered to and this is twice the "optimum" value, for example, then the flight path time delay would be half the value of $T\theta 2$. As the latter would be quite typically about 0.5 seconds in LAHS conditions the

resulting attitude drop back would only be about 0.25 seconds, which as shown above can be satisfactory for precision handling. For a high order FCS the application of Fig. 8 is somewhat indirect, indicating rather the "optimum" normal acceleration time response via Fig. 3 or 4. For such an aircraft the more direct approach of specific shaping of the attitude response would of course be adopted instead.

From the combination of the facts that the 8785 frequency is proportional to speed (Table 1) and the flight path time delay is inversely proportional to frequency (given constant manoeuvre margin and damping), and hence that this time delay is inversely proportional to speed, it will be observed that the path distance represented by the delay is constant. In effect this reveals that the flight path response bandwidth could be considered to be constant and independent of speed, which may be of relevance to close-in air-to-air combat. If this is the case then this result is compatible with the concept of a fixed attitude frequency response bandwidth independent of speed also, a subject discussed here and also in Ref. 3.

4. ATTITUDE FREQUENCY DOMAIN RESPONSE (UP-AND-AWAY FLIGHT)

In the late 1950's and early 1960's the systems analysis approach to the pilot-vehicle closed loop control in tracking tasks was widely investigated and reported. Ref. 4 was based on such methods and contains many references to the literature. The more recent "Neal and Smith criterion" is based upon this method also. The pilot is assumed to act as a linear servo element with gain, time delay, and phase lead and phase lag equalisation to produce satisfactory tracking performance. "Tracking" of course covers a wide range of tasks, from controlling aircraft attitude in turbulence to actual tracking of a target. In most cases the pilot is assumed to respond to the stimulus of the error between the desired attitude and the actual one, an error assumed to be random in appearance in which the pilot is unable to distinguish the effects of his control inputs from external effects. This is the pure compensatory tracking situation and represents the other extreme of the pilot task from the precognitive time response situation.

Much work was done in attempting to predict pilot opinion from such analyses, though this does not seem to have been followed in more recent years with the exception of the Neal and Smith criterion. With the advent of the computing power potential of digital FBW it is now much more useful to the FCS designer to attempt to shape the aircraft frequency response into a form known to be attractive to the pilot, with which he can perform both well and easily and hence will result in a good opinion rating. The pilot model which achieves this aim is well known to be the simple gain and time delay, the latter always being present in random error tracking. It is possible to define an envelope of aircraft attitude response which is very "robust", in the sense that the pilot can achieve good closed loop control witha wide range of gain and delay only. In this approach it is unnecessary to define a pilot-vehicle bandwidth since he has a wide choice according to the needs of the task.

The classical aircraft dynamics which have always been shown to achieve the best ratings in simple tracking experiments are a pure gain pitch rate response and the resulting attitude response of K/S. Real aircraft have inertia, control power limits, and pilots who dislike excessive pitch acceleration, and can only be represented by this model at low frequencies. These attitude responses are indicated in Fig. 9 using the Nichols' chart form on which open loop and closed loop responses are related. Because of this facility these charts are often more useful to the FCS control law designer than the more usual Bode plots, even where no pilot model is being added to the aircraft response. In Fig. 9 a pilot gain and delay model is added to a pure low order attitude response to show good closed loop performance with negligible droop or resonance. The gain is chosen to give the pilot-vehicle open loop crossover frequency of 0.3 Hz and a small delay typical of simulation results is selected. A K/S response is included for comparison with the aircraft response with the same crossover frequency.

This basic pilot model assumption underlies the aircraft response boundary limits used as a design criterion. The crossover frequency typifies the upper end of the 1 to 2 rad/sec. range and the 0.2 second time delay typifies the pilot delay noted generally in the literature in simulation experiments. Ref. 5 measured the difference between flight and simulation and showed that, while the lead or lag equalisation did not change, the pilot gain was lower and the time delay was larger in flight.

Choice of these values therefore represents an upper limit on pilot performance in the definition of aircraft response boundaries. The choice of frequency in Hz rather than radian/second is deliberate since the pilot sees frequency behaviour in terms of its period or cycles per second, and this serves to present a more obvious view of the effect of such boundaries.

Fig. 10 shows optimum aircraft pitch attitude response boundaries for precision control tasks, in which the crossover frequency of 0.3 Hz is inherently assumed. To apply this criterion the aircraft attitude response to the stick input is plotted so that its open loop amplitude at 0.3 Hz is 0dB. The phase difference between this and the 0dB closed loop line represents the allowable pilot phase lag for optimum tracking. If this criterion is satisfied, all the pilot phase lag can be attributed to his time delay and no further equalisation is required from him. If the pilot chooses a lower crossover frequency the allowable lag increases, and he can adopt a larger time delay without departing from a good closed loop performance.

These boundaries do not represent overall Level 1 limits. Depending on the task, responses outside them can be very satisfactory. General characteristics associated with areas outside the boundaries are indicated, and were derived from correlation with comment data from Ref. 1 and up-and-away flight configurations. While the boundaries represent the small number of Ref. 1 cases which were optimum for all tasks, some cases attained Level 1 ratings for flight refuelling despite bobble severe enough to degrade pitch tracking to Level 2, and the best flight refuelling case ("really excellent") lay within the boundary confines but was still a poor Level 2 for tracking because of a subtle blend of large attenuation at higher frequencies and unbalanced initial-to-final time response. This case had excellent flight path control with small delay, provided that aggressive attitude control was not attempted. More generally, excessive attenuation at all frequencies is associated with sluggish unpredictable flight path control also.

5. PILOT INDUCED OSCILLATIONS

PIO tendencies were noted in Ref. 1 for several cases which were related to low damping or resonance of the pilot-aircraft closed loop combination. Apparently these could readily be stopped by abandoning the task or reducing the pilot gain, and flight safety did not seem to be threatened by sustained "locked-in" oscillations. This result is probably general provided that the response remains linear and predictable although unsatisfactory or even unacceptable for performance achievement. Ref. 4 discusses the probable causes of a number of PIO events and concludes that for aircraft with pure low order characteristics, PIO is impossible. Where the gain margin suddenly reduces by a significant amount, however, a sustained PIO may result due to the pilot's inability to reduce gain quickly enough. This has been caused in the past by spring-bobweight pitch feel systems, for example, but a more likely cause with FBW would by actuator saturation effects, more easily provoked than before because of high forward path gains. Dramatic changes in gain and frequency can result from increasing amplitude inputs which outstrip the pilot's ability to adapt. It has to be accepted by weight-conscious systems managers than an aircraft designed for "Active Control Technology" really does require active controls!

Ref. 4 proposed a PIO criterion, given in Fig. 7, which was applicable to low order aircraft. It results from the condition for the open loop frequency and damping at which pilot-aircraft closed loop damping tends to zero, when the pilot is responding without phase lag to a sustained attitude oscillation at the frequency where the aircraft phase lag is 180 degrees. While such a condition requires very unconventional aerodynamics in principle, it is certainly possible to achieve it with variable stability aircraft and simulators. In such cases it is invariably seen to result in poor opinion ratings and examples of this are given in Ref. 3, Ref. 4, Ref. 6 and Ref. 7, though they are not always identified as such.

The concept of the "synchronous pilot" who can eliminate phase lag from his inputs in a sustained PIO because of the predictable response is a very powerful tool in designing to avoid PIO. Such behaviour can be seen in almost any overcontrol situation involving attitude control, which comprise the great majority of PIO's. It may not always be absolutely exact but it is close enough to permit consideration of only the attitude response at 180 degrees phase lag without attempting to predict how the pilot will actually get into a PIO. If the oscillation is sustained for any length of time the pilot can sometimes be seen to advance his phase by as much as 90 degrees, but at the beginning he will drop immediately into the synchronous mode.

No hard and fast PIO avoidance criteria can be given with assurance for up-and-away flight but if the attitude response remains reasonably linear with amplitude, follows the criterion boundaries in Fig. 10 reasonably closely, and crosses 180 degrees phase lag at a high frequency beyond the range of significant pilot activity (say above 1.5 Hz), then PIO in up-and-away flight is remote. It is also certain to be avoided regardless of handling ratings if at the 180 degrees phase lag frequency the application of stop-to-stop stick inputs results in small attitude response, say $\pm$ 2 or 3 degrees. Exactly similar considerations apply to bank attitude PIO, and the same rules should be applied here also.

While these criteria are intended to permit design of control laws without pilot models, not all projects will offer the flexibility of a FBW system, and it is always of value to acquire some appreciation of the pilots' capabilities and limitations. Ref. 7 discusses these as derived from a fixed base simulation and covers a very wide range of characteristics such as linearity, non-linear remnant, output frequency range, the poor correlation of pilot opinion and tracking performance, the good correlation of tracking performance with output linearity, and so on. Fig. 11 illustrates the pilot's ability to equalise four widely different attitude responses from Ref. 7. In three of them his output is largely linear and he is able to reproduce surprisingly well the K/S aircraft response in the 0.1 to 0.2 Hz frequency range where his output is most significant. In the fourth case he is quite unable to achieve this and the linear model represents only 50% of his output while the other 50% is non-linear and results in a rather frantic switching technique. The boundary found in these experiments between linear and significantly non-linear behaviour, which always results in poor opinion can be shown to lie along the line of $2\zeta\omega n = 1/T\theta 2$, the Ref. 4 PIO criterion.

It is therefore possible to construct a set of worst case never-exceed low-order boundaries for different damping levels at any value of $1/T\theta 2$, in Bode form to separate them clearly. Fig. 12 shows these for $1/T\theta 2$ of one radian per second. For any other value the boundary shapes remain identical but the frequency points are multipled by $1/T\theta 2$. Note in all cases the low frequencies at which the normal maximum phase lag value of 180 degrees is reached, and indeed exceeded despite the absence of control system dynamics in Case 4 of Fig.11 which lies beyond these boundaries. It is of interest to note that such combinations of low frequency and low damping can be shown to satisfy 8785 requirements, for example CAT.B, with perfectly reasonable values of wing loading and lift-slope. As discussed earlier, however, it would require unusual aerodynamics because low frequency is associated normally with low manoeuvre margin and high damping. Application of these boundaries to a high order system, which could very well display large phase lags at low frequencies if incorrectly designed, has to be qualitative in nature. This is because the low order steep attenuation near 180 degrees lag is usually replaced in such cases by a rapid traverse in phase lag across 180 degrees finishing up eventually with much higher lags, probably resulting in worse PIO tendencies.

The part played by normal acceleration in PIO's can be no more than qualitative in the sense of increasing the pilot's concern about the structural safety of his aircraft. Ref. 8 clearly demonstrates the inadequacy of the human linear acceleration sensor, the otolith, to provide accurate feedback. Humans are indeed able to sense acceleration as small as 0.01g, but the latency time before a steady g of this level is detected is about five seconds. The dynamic response has large attenuation and phase lag at frequencies beyond 0.25 Hz, and its actual gain seems somewhat indeterminate.

While it is certainly difficult to provoke PIO in fixed base simulators it is by no means impossible, and the resulting pilot behaviour is indistinguishable from typical flight records. The "synchronous pilot" response to attitude is very easy to demonstrate also. The fact that fixed base simulation is an unreliable guide to PIO seems certainly due to absence of attitude acceleration cues from the semicircular canals, probably poor visual display dynamics in the past, and in particular poor cues representing close approach

to the ground in landings. Lateral PIO's in bank attitude have the same observed synchronous pilot behaviour as pitch PIO's and obviously can have no input from a linear acceleration. It has always proved sufficient to treat the "rapid PIO" as a single loop attitude response.

6. THE LANDING APPROACH

This task deserves separate treatment because it can never be avoided and has sometimes revealed marked tendencies to PIO below 50 foot altitude even where this is not detectable in circuit flying or in simulations. It is certainly a multi-loop task in attitude, flight path and speed, and poor achieved performance is penalised by a bad and possibly damaging landing. The drag and thrust characteristics which determine whether the aircraft is a "floater" or a "sinker" at touchdown are not discussed here and have not usually been the direct concern of the FCS designer, although this will change with the introduction of integrated engine and flight control management for special configurations. In the following paragraphs it will be shown that satisfactory pitch attitude and flight path response characteristics can be defined for low or high order FCS systems without difficulty.

Ref. 9 proposed a mechanism for the initiation of landing PIO which has been successfully extended to include all the cases for Ref. 2. The PIO starts when the pilot begins to perceive the attitude oscillation in the stick pumping activity in the flare manoeuvre. The latter, noted first in Ref. 10 seems to be a subconscious testing of the pitch control by pumping at a frequency where pitch acceleration is nearly in phase with the stick, and at an amplitude of about 6 deg/sec^2. In consequence the attitude oscillation is 180 degrees out of phase with the stick, so that the pilot is already "set up" for a PIO if he detects it consciously.

In the true low-order aircraft there is substantial attitude response attenuation before the pumping frequency is reached, itself considerably higher than the bandwidth used by the pilot, and the phase lag only gradually approaches or exceeds 180 degrees so that the frequency choice is not critical. The high order FCS can transform this so that the frequency is low, giving a large attitude response at the constant acceleration level, and the phase lag may rapidly pass through and far beyond 180 degrees. These factors are sufficient to switch on the latent "synchronous pilot" into a PIO.

Fig. 13 shows an open loop criterion which incorporates all the relevant Ref. 2 cases. Neither the attenuation (effectively a gain margin of a sort) nor the amplitude alone are a sufficiently accurate metric but combined as shown they form a clear indication of potential PIO. Although the rating is generally predicted only to an accuracy of 3, all serious PIO cases with an 8 to 10 rating are unambiguously identified with large criterion values and all very highly rated cases with small values. However, a further factor is that no PIO can occur if the response amplitude at 180 degree phase lag is so low that even stop-to-stop stick inputs generate only a small oscillation e.g. 2 or 3 degrees attitude, as noted earlier.

Although a great deal can be inferred about touchdown behaviour from a very small slice of the attitude frequency response, overall open loop boundaries are necessary to complete the design process. Those in Fig. 14 were derived from the Ref. 2 cases. They effectively divide the response regions into two at a bandwidth limit set by the frequency at which phase lag is 120 degrees. Above this frequency the PIO tendency at touchdown is determined as in Fig. 13, and reponses satisfying the boundaries automatically achieve low PIO criterion values. Below the bandwidth frequency the responses useful for flight path control are defined. The satisfactory range of bandwidth limit frequencies lies between about 0.25 and 0.5 Hz.

Briefly discussed in Section 3 above, the Ref. 2 cases provide data on satisfactory landing approach time responses. Large values of attitude drop back, at least up to 1.0 second, which would be wholly unacceptable for precision tracking, seem to be acceptable with some constraint on initial acceleration and pitch rate overshoot ratio. This is probably because attitude is not controlled agressively in the landing. The flight path time delay should not exceed 2.0 seconds for satisfactory approach control and should certainly be less for optimum touchdown control.

It will be observed from the above results that satisfactory approach handling can be achieved with values of $T\theta 2$ as large as 3.0 seconds. For quite typical wing loading and lift slope this could be associated with normal acceleration per radian of 2.0 for example, well below the 8785 minimum. For equally typical flare responses of 1.0 deg/second pitch rate or 0.1g, such a configuration needs only an increase of 3.0 degrees angle of attack to achieve this. Stall margins do not impose a limit so long as conventional speed margins apply. Such considerations strongly support the results in Ref. 11 which conluded also that minimum approach speed should be set by $T\theta 2$.

It may also be observed that at the 8785 Level 1 minimum frequency and maximum damping the flight path time delay is 3.0 seconds and the time response takes more than 10.0 seconds to settle after an input, which are certainly not satisfactory values. For a damping of 0.7 however, these values reduce to a satisfactory 1.9 and 4.0 seconds.

In all cases where such landing approach criteria have been tested they have successfully described the aircraft behaviour, commencing in 1972 with use of the stick pumping criterion (Ref. 10) to predict prior to first flight the final approach control activity at engine idle conditions, through the initial criterion development described in Ref. 9, followed by expansion into the present forms described here using the Ref. 2 data with recent application to published characteristics and comment ratings of the F14, YF12, Space Shuttle and B1. It is especially interesting to note that aircraft as large as the latter two are described as sluggish or satisfactory by exactly the same criteria as small Class 4 aircraft.

The final proof of validity has been demonstrated by the digital FBW Jaguar, with probably the first FCS designed with a specific process for avoiding landing PIO as well as providing excellent approach and landing characteristics, in both the pitch and (by similar methods) the roll axes. The handling has turned out as predicted, down to the precise stick pumping amplitude and frequency, and remains excellent in severe crosswind and turbulence conditions. It is believed with confidence that application of these criteria to other new designs will ensure similar handling despite the inability to detect PIO reliably on fixed base simulators.

7. DELAYS AND PHASE LAG

Time delays or transport delays essentially do not occur in analogue FCS, whether high order or not. They occur in digital FCS, of course, but it is rather unlikely, when limited to values essential to attain system stablity, especially so for increasingly unstable airframes of the future, that they will have any discernible effect on handling. As noted in the PIO discussion, discontinuities and nonlinearities due to actuation rate and acceleration limits are potentially far more serious in a high gain FBW system and require careful treatment.

Phase lags are inseparable from analogue or digital control systems and excessive values have been the root cause of most FBW handling problems. However, it is a mistake to believe that specification of in-flight stick to control surface phase lag is either necessary or sufficient to prevent such problems, as two examples will demonstrate.

- In a low speed landing configuration where a low short period natural frequency has been augmented to a higher value, the airframe will of couse lag the control surface by large phase angles at frequencies used in control inputs. It is therefore essential to ensure phase lead in the surface activity to obtain acceptable airframe phase lags. Meeting a phase lag requirement only would probably ensure PIO.

- A very high short period natural frequency (16.0 r/s) configuration was simulated in Ref. 1 Case 8, which would of course have required surface phase lead in the NT33 airframe. The optimum version, 8D, had a stick lag prefilter with 0.3 seconds time constant, which has about 60 degrees phase lag at 1.0 Hz. If a "real" airframe had had such a high frequency it would have required the same lag prefilter. In this case meeting a small phase lag requirement would ensure an excessively abrupt pitch response and prevent precision control achievement.

While low actuation phase lags are necessary for FBW systems, which could legitimately require some specification, control surface behaviour in flight is not uniquely definable except as the indirect result of achieving satisfactory attitude or flight path response.

The latter can be completely described and specified by the criteria presented in this paper without reference to the control surfaces.

No definition has been seen so far which adequately defines the difference between the low order and high order FCS. It is suggested that what matters to the pilot is not the transfer function order but the apparent "order" of the response which he sees. This can be defined qualitatively by the maximum possible phase lag of any of these responses, for example zero degrees for pitch acceleration, 90 degrees for pitch rate, 180 degrees for pitch attitude, and 270 degrees for flightpath angle, for the pure low order aircraft. Long experience has shown that the addition of moderate actuation phase lags need not alter the essential low order characteristics so far as the pilot can observe them. It is also well established that good handling qualities are confined to regions within this broad definition of low order responses.

Hence an overriding consideration for high order FCS design should be an attempt to contain phase lags to values no greater than the above plus say an extra 30 degrees for all frequencies below 1.5 Hz or preferably even 2.0 Hz.

8. CONTROLLER CHARACTERISTICS AND STICK FORCES

It is well known that rigid centre and sidesticks have been flown in a number of programmes. More satisfactory characteristics have also been shown to result from introduction of some motion of sidesticks and considerable research has been conducted in recent years on the Calspan NT33 to determine the required force-displacement gradients.

Centre sticks will probably continue to be widely used for many years. The lack of clear guidance to optimum force-displacement gradients probably reflects the ability of most pilots to perform most tasks acceptably with a wide range of gradients. (See for example Ref. 12). The exceptionally demanding LAHS environment was studied in a g-seat simulator experiment reported in Ref. 13, in which great care appears to have been taken to validate results against flight and to involve an unusually large number of pilots. This did produce clear guidance which has subsequently correlated well with a series of aircraft designed for the LAHS role.

It is worth noting that the inadvertent pitch stick displacement caused by arm jostling in turbulence was only weakly related to feel stiffness. Consequently the effective command disturbances tended to increase as the feel stiffness increased, roughly in the proportion of stiffness ratio to the power of 0.75. Pilot overall rating was found to correlate with feel stiffness rather than force or displacement per g, but satisfactory values lay between 3 and 25 lb/in. (0.5 to 4.4 N/mm). However, the simulated terrain following performance achieved minimum height errors for moderate stiffness around 7 to 12 lb/in (1.2 to 2.1 N/mm) with light stick force per g. Hence aircraft intended for LAHS operation should have a pitch feel stiffness in this region, which will be satisfactory also for other tasks.

Ref. 9 discussed the 8785 stick force per g boundaries and noted that the minimum value should increase at low speeds rather than remain constant as permitted. This view can be further substantiated by consideration of the following:-

- The Ref. 1 data showed the need for the same attitude bandwidth for flight refuelling as for combat at higher speed, a quite well established value from a wide range of experiments. Ref. 14 obtained satisfactory rating for a range of pilot gain per degree of attitude error between 3 and 12 lb/degree (13 to 53 N/degree), which agrees well with the 15 to 25 N/degree noted to be within satisfactory limits in Ref. 9. A constant value of this gain implies an inverse function of speed for stick force per g. At higher speeds where a constant stick force per g is desirable for compatibility with structural limits the consequential reduction in attitude response gain with increasing speed is

probably also favourable for LAHS and it can be contained within satisfactory limits in any case.

- Analysis of the stick command gains selected by the Ref. 2 pilots shows poor correlation with stick force per g. Chosen gains were 6 to 60 lb/g (27 to 270 N/g) with a general trend to low values for low bandwidth configurations, but Level 1 cases lay between 14 and 38 lb/g (62 to 170 N/g).

In fact the Ref. 2 results indicate that pilots prefer stick force gains related to pitch attitude response for landing approach. Fig. 15 shows the striking correlation of their chosen gains with attitude response at the bandwidth frequency where the phase lag is 120 degrees. The higher response of the group of low damping cases simply reflects pilots' ability to distinguish and tolerate the open loop response resonance, usually near its peak at this frequency, provided that the phase lag and attenuation at higher frequencies resemble low order behaviour.

Ref. 7 proposed an attitude tracking sensitivity criterion which quite successfully matched the ratings for the 40 simulated configurations. This added the proposed opinion rating penalties for sluggish response at 1 rad/sec, excessive response at 5 rad/sec, and excessive phase lag at 0.2 deg/lb response amplitude. The latter represents the phase margin for the "pure gain pilot" at 5.0 lb/deg. gain. The three corner points beyond which rating decrements were indicated are shown in Fig. 16. This accurately differentiates the characteristics of two FCS standards flown in the development phase of a recent combat aircraft and a brief survey of some other cases suggests that it could be the basis of a very useful criterion for high order FCS. It is proposed that a wide survey of available flight data should be made to establish valid limits for general application. Note however that there appear to be national preferences for stick force levels which may need to be allowed for. UK pilots for example generally seem to consider 8785 stick forces on the heavy side.

9. CONCLUDING REMARKS

The use of digital computing has permitted the FCS designer to modify the handling qualities of aircraft to levels never before experienced. The first lesson to be learned from almost universal experience is that significant departure from the classic low order aircraft dynamic response characteristics will probably lead to poor handling qualities.

The second lesson is that the intensive study and analysis of handling qualities conducted in the 1950's and 1960's provides most of the information necessary to design the modern high order FCS control laws. This paper has presented a variety of methods for interpreting the intrinsic pitch behaviour of classic airframe response in ways not solely dependent upon a definition of low order parameters which may no longer exist meaningfully. From a study of in-flight experimental data a range of satisfactory responses related to flight phase has also been identified.

The importance of time response criteria is strongly emphasised, each individual facet having an influence on rating which can often be adjusted to advantage. Some of these facets can be qualitatively identified from frequency responses but the latter should never be adjusted without consideration of the time repsonse changes.

The criteria described are coming into routine use and have already proved easy to use, presenting to the engineer a clear view of the handling as seen by the pilot and thus enhancing the communication between them.

The power of the combined application of time and frequency response criteria is such that a good appreciation of the handling qualities of a configuration can be obtained almost on sight of the calculated response. Often the required adjustment for final optimisation can be defined by a simple inspection process.

The gains achieved in the 1970's have already been considerable. The foundations for the future development of CTOL aircraft in unconventional directions have been firmly laid by the fresh insights into handling qualities resulting from application of FBW to conventional aircraft.

REFERENCES

1. Boothe, E.M., Chen, R.T.N., Chalk, C.R., Calspan Corp.,
A Two-Phase Investigation of Longitudinal Flying Qualities for Fighters
April 1974, AFFDL-TR-74-9

2. Smith, R.E., Calspan ATC, Effects of Control System Dynamics on Fighter Approach and Landing Longitudinal Flying Qualities
March 1978, AFFDL-TR-78-122

3. Hoh, R.H., et al, STI., Hodgkinson, J., MacDonnell Aircraft Co.,
Proposed MIL Handbook - Handling Qualities of Piloted Airplanes,
December 1981, AFWAL-TR-82-

4. Ashkenas, I.L., Jex, H.R., McRuer, D.T., STI, Pilot Induced Oscillations: Their Cause and Analysis
June 1964, STI TR-239-2

5. Seckel, E., Hall, I.A.M., Princeton University, McRuer, D.T., Weir, D.H., Control Specialists Inc.,
Human Pilot Dynamic Response
August 1958 WADC TR 57-520

6. Eney, J.A., NADC, Flight Evaluation of Power Approach Short Period Dynamics in the Variable Stability NF-8D,
August 1969, AIAA Paper No. 69-896

7. Hall, I.A.M., Study of the Human Pilot as a Servo-Element, Jnl. of the R.Ae. S., Vol. 67, No. 630
June 1963 pp351-360

8. Meiry, J.L. M.I.T., The Vestibular System and Human Dynamic Space Orientation,
October 1966 NASA CR-628

9. Gibson, J.C., British Aerospace, Flying Qualities and the Fly-By-Wire Aeroplane,
September, 1978 AGARD-CP-260

10. Bihrle, W., A Handling Qualities Theory for Precise Flight Path Control,
June 1966 AFFDL-TR-65-198

11. A'Harrah, R.C., Lockenour, J.L., North American Rockwell, Approach Flying Qualities - Another Chapter,
August, 1969 AIAA Paper No. 69-895

12. Abramovitz, M., Van Dyke, R.D., NACA, The Effect of Stick Force Gradient and Stick Gearing on the Tracking Accuracy of a Fighter Airplane,
September, 1954 NACA RM A54I28

13. A'Harrah, R.C., North American Aviation, Low Altitude High Speed Handling and Riding Qualities, Jnl. Of Aircraft, Vol. 1, No. 1
January-February 1964, pp. 32-40

14. Sadoff, McFadden, Heinle, NASA, A Study of Longitudinal Control Problems at Low and Negative Damping and Stability with Emphasis on Effects of Motion Cues
January 1961, NASA TN D-348

	RANGE OF PHASE ANGLES
● $\frac{n_{z\,CG}}{\delta}(S) = K_{n_z}\left[\frac{1}{\Delta}\right]$	$0 \rightarrow -180$
$\frac{\theta}{\delta}(S) = K_q\left[\frac{1+T_{\theta 2}S}{S\Delta}\right]$	$-90 \rightarrow -180$
$\frac{q}{\delta}(S) = K_q\left[\frac{1+T_{\theta 2}S}{\Delta}\right]$	$0 \rightarrow -90$
$\frac{\dot{q}}{\delta}(S) = K_q\left[\frac{S(1+T_{\theta 2}S)}{\Delta}\right]$	$90 \rightarrow 0$
$\frac{\gamma}{\delta}(S) = K_q\left[\frac{1}{S\Delta}\right]$	$-90 \rightarrow -270$

$$\Delta = 1 + \frac{2\zeta}{W_n}S + \frac{S^2}{W_n^2}$$

● $$\alpha_{ss} = T_{\theta 2}\, q$$

● $$n_{z_\alpha} = \frac{V}{g}\frac{1}{T_{\theta 2}}$$

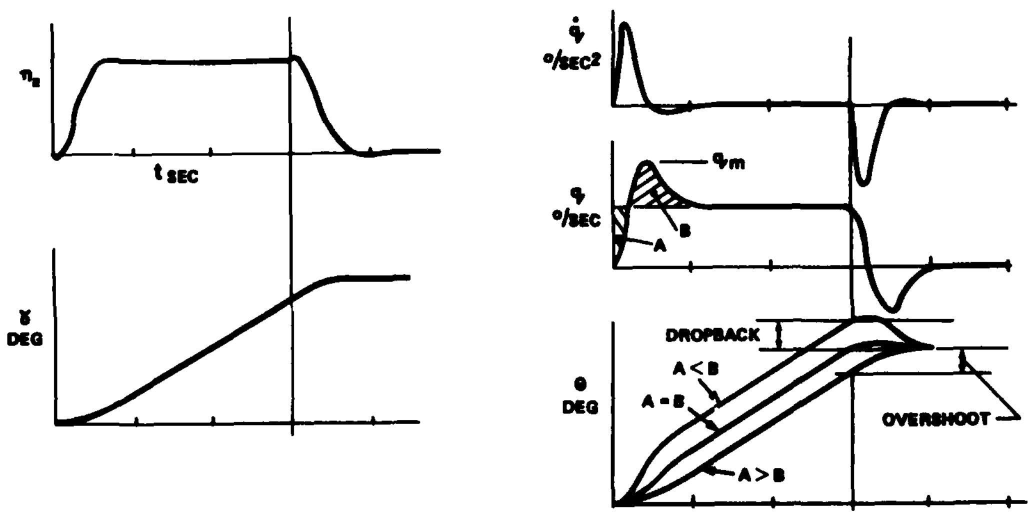
● 8785 CAT. A L.I $W_{n\,(MIN)} = 0{\cdot}55\sqrt{n_{z_\alpha}}$

$W_{n\,(MAX)} = 1{\cdot}95\sqrt{n_{z_\alpha}}$

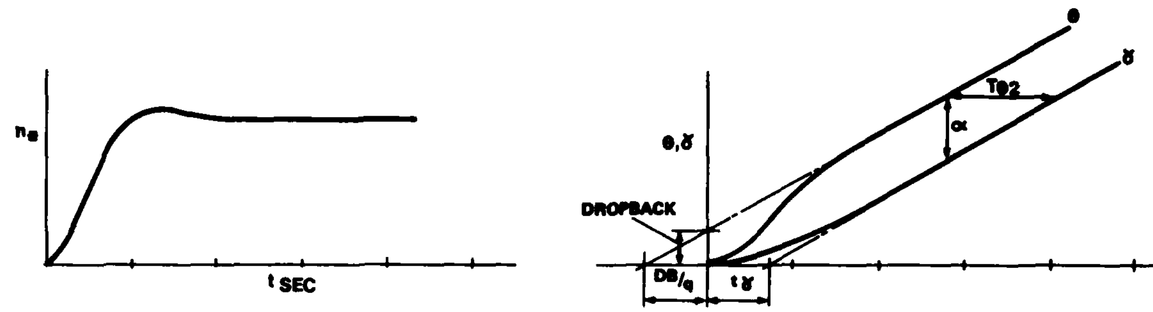
●

n_z RESPONSE	$\zeta =$ 0·7	1·0	1·3
$t_{\dot{n}_z}$	$3{\cdot}5/W_n$	$6{\cdot}0/W_n$	$9{\cdot}0/W_n$
t_γ	$1{\cdot}6/W_n$	$2{\cdot}1/W_n$	$2{\cdot}6/W_n$

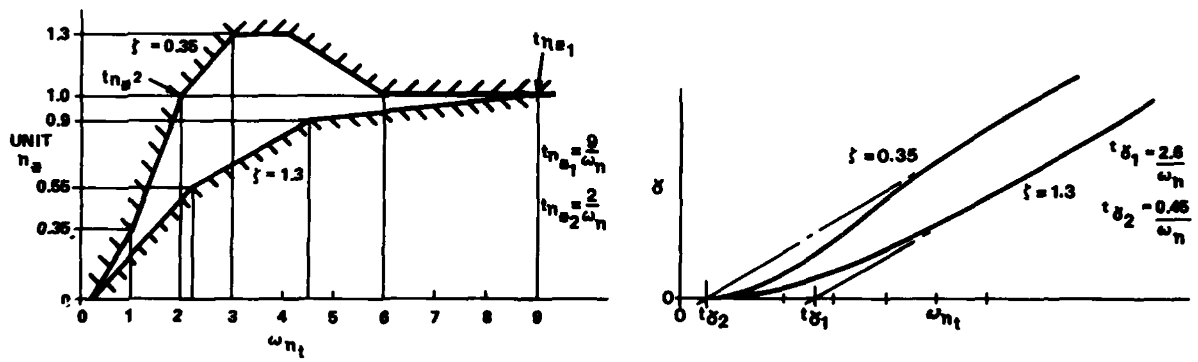
SIMPLIFIED RELATIONSHIPS Table 1



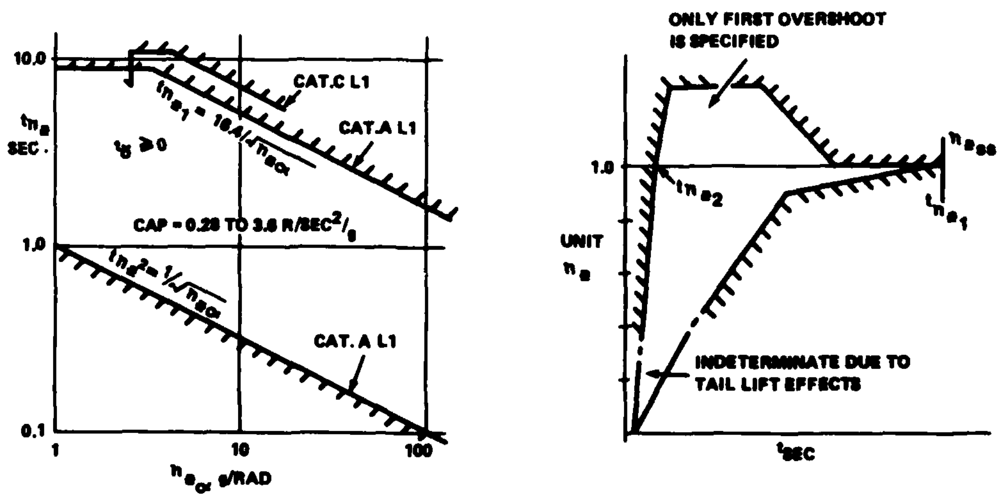
TIME RESPONSE ANALYSIS FEATURES Fig. 1



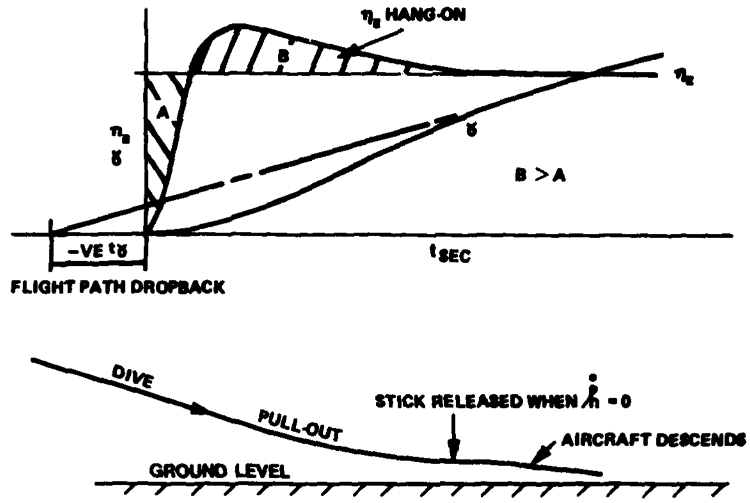
TIME RESPONSE RELATIONSHIPS Fig. 2



TIME RESPONSE BOUNDARIES Fig.3

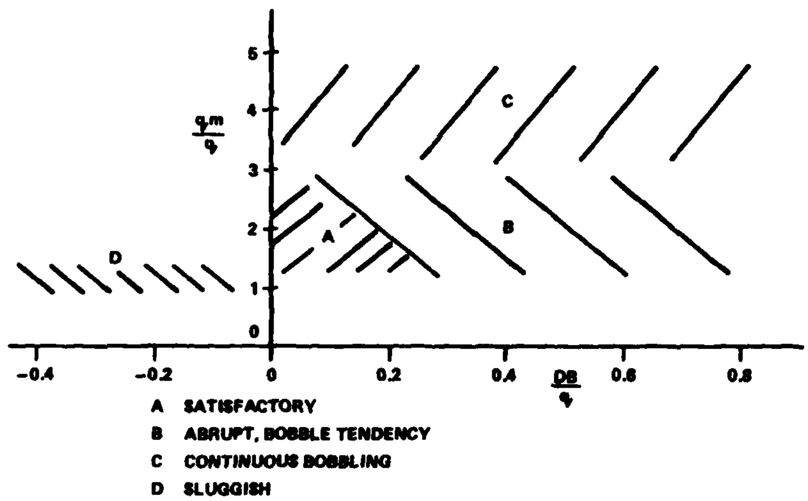


EQUIVALENT W_n & ζ BOUNDARIES Fig.4



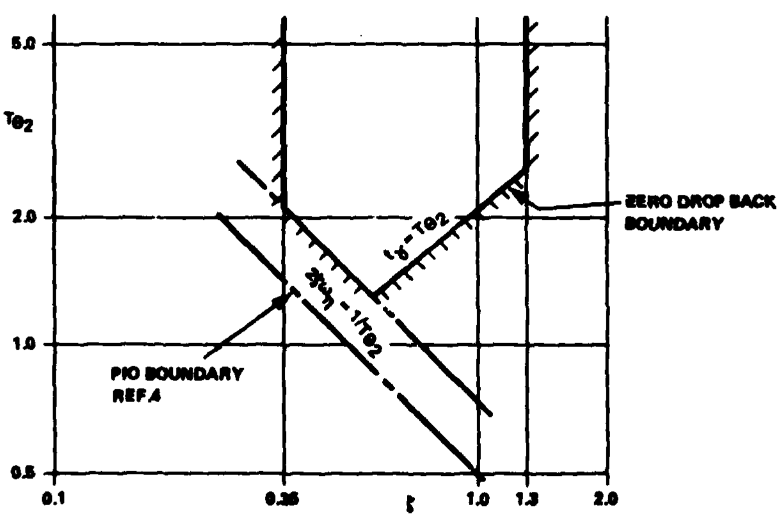
n_z HANG-ON EFFECTS

Fig.5



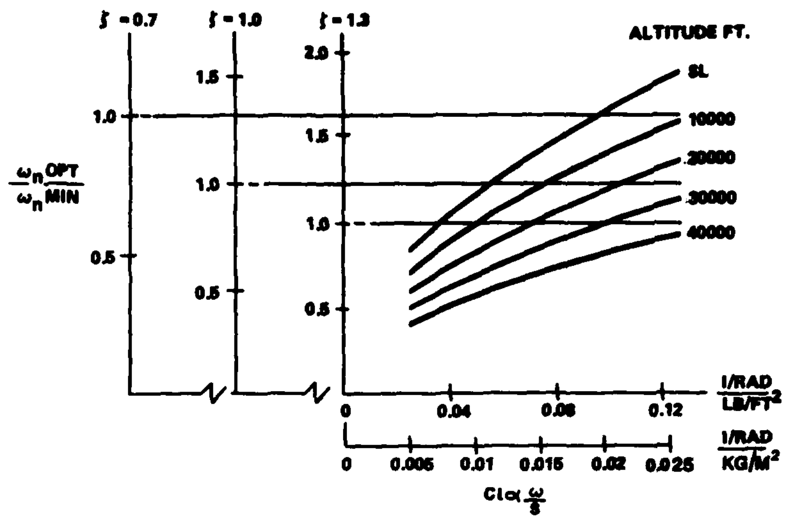
PRECISION TRACKING : $q_s \sim \theta$ TRENDS

Fig.6

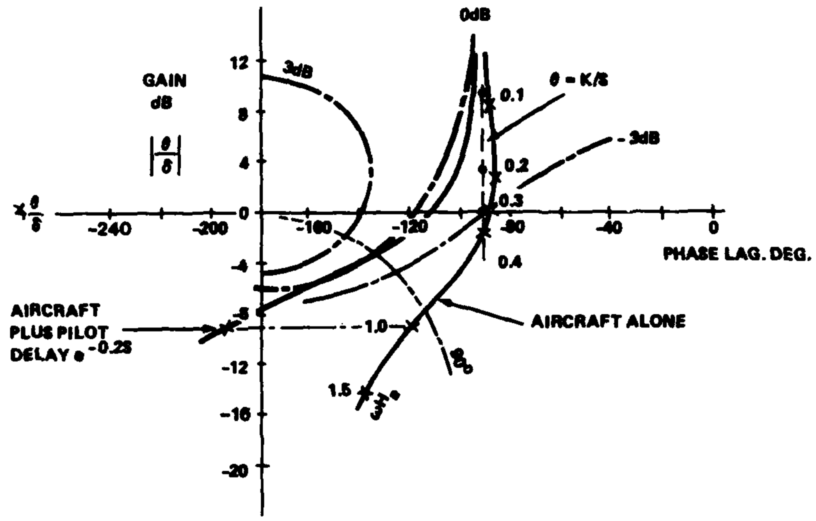


COMBINED ω_n, ζ, T_{θ_2} BOUNDARIES

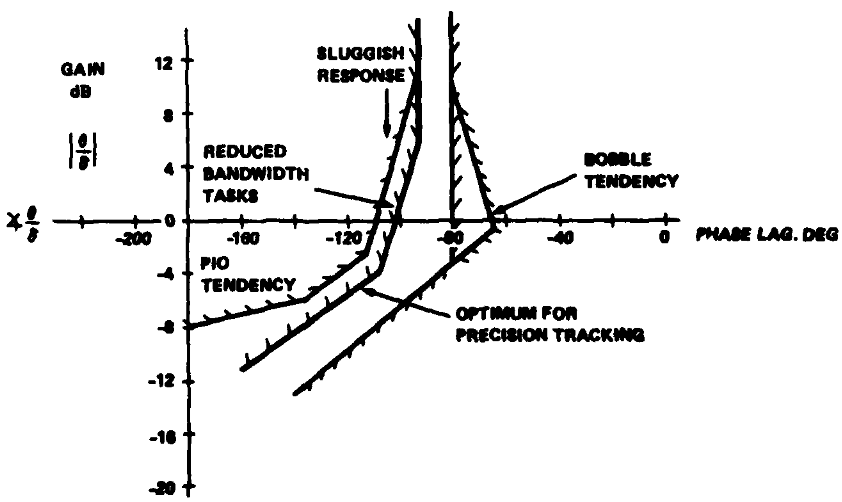
Fig.7



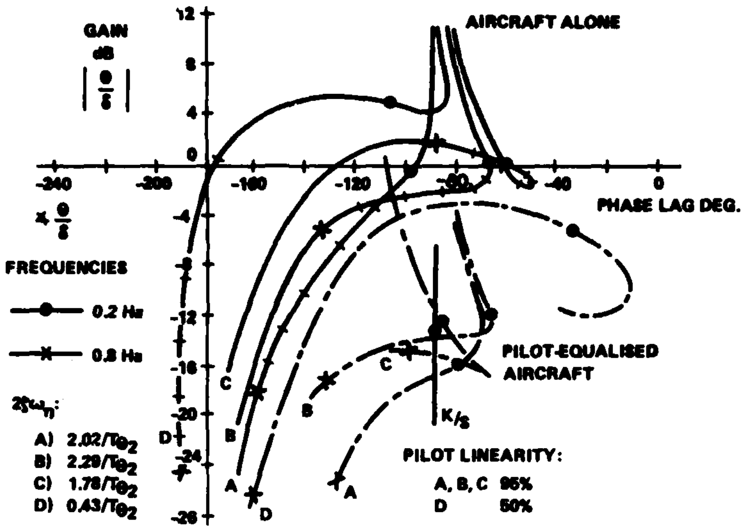
RATIO OF ZERO DROPBACK ωn TO 8785 MINIMUM **Fig.8**



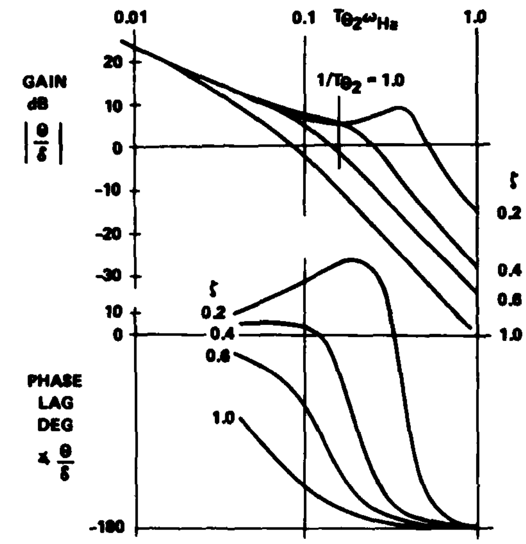
LOW ORDER PILOT–AIRCRAFT ATTITUDE FREQUENCY RESPONSE **Fig.9**



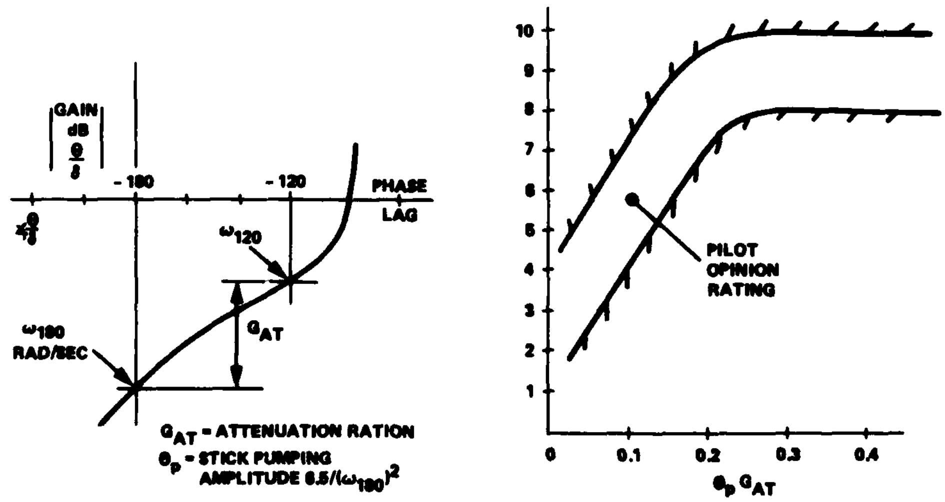
ATTITUDE BOUNDARIES FOR 0·3 Hz CROSSOVER FREQUENCY **Fig.10**



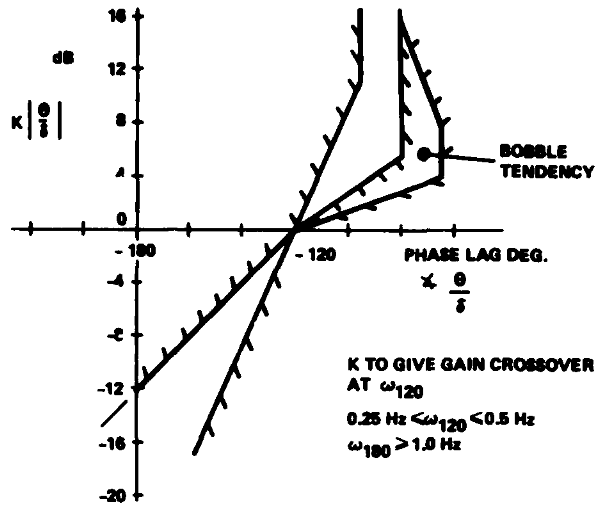
Ref. 7 PILOT EQUALISATION CAPABILITIES Fig.11



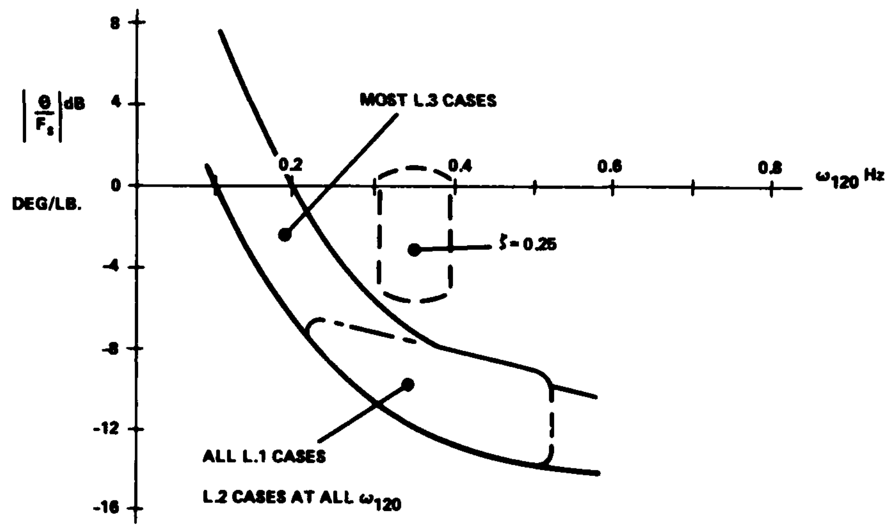
$2\zeta\omega n = 1/T\theta_2$ BOUNDARIES Fig.12



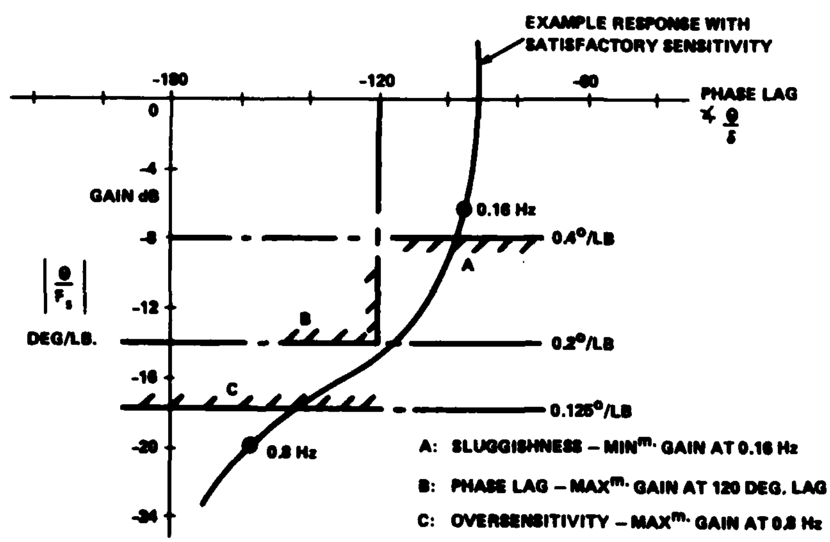
LANDING P10 CRITERION Fig.13



LANDING APPROACH OPTIMUM ATTITUDE RESPONSE Fig.14



Ref.2 PILOT SELECTED STICK FORCES Fig.15



Ref.7 PITCH TRACKING SENSITIVITY Fig.16

GAIN AND PHASE MARGIN AS A BASIS OF LONGITUDINAL FLYING QUALITIES EVALUATION

by
Dr. W. Röger and H. Beh
MESSERSCHMITT-BÖLKOW-BLOHM GMBH
Aircraft Division
8000 München 80, Postfach
Germany

SUMMARY

A criterion is presented that allows an evaluation of the longitudinal maneuvering characteristics of modern fighter aircraft. The required parameters are the 'gain margin' and the 'phase margin' of the frequency response characteristic of pitch attitude to control force. This criterion permits an evaluation of the dynamic characteristics as well as the steady-state and assumes that the pilot is always looking for a compromise between stability and response time. The criterion enables the estimation of PIO-tendencies, gives a survey about the influence of different parameters like time delay, lead time, natural frequency, damping ratio and the ratio of control force per normal load factor. The design of an advanced flight control system as an example illustrates the conformity of this criterion with MIL-F-8785.

1. INTRODUCTION

The classical requirements of the longitudinal flying qualities are given in MIL-F-8785C (1). In the case of longitudinal maneuvering characteristics these specifications evaluate the handling qualities on the basis of a low order system whereby airplanes presenting a high order system must be reduced by application of a matching method. This report represents a criterion for the evaluation of the longitudinal maneuvering characteristics that is independent of the order of the system. This criterion integrates the evaluation of the dynamic characteristics as well as the steady-state. It has been developed for airplanes of Class IV and for flight phases of Category A as defined in (1). Fundamentally the criterion itself is applicable for other classes of airplanes and other flight phases, but in this cases the limits must be changed and validated.

It is assumed here that for flight phases of Category A, like air-to-air combat or close formation flying, the pilot closes the attitude control loop. Therefore this loop has been selected for the development of the criterion.

For the evaluation of the maneuvering characteristics the frequency response characteristic of the pitch attitude/control force transfer function is chosen. The pilot rating applies to this part of the control loop which includes subsystems like control stick, linkage, flight control system, sensor, actuator and the airframe itself. Together with an assumed unity gain for the pilot, the open loop attitude control characteristic is fully realized. Consideration of the open loop characteristic gives the possiblity of applying control theory. The characteristic parameters for the design of a control loop with the aid of the frequency response characteristic of the open loop are the 'gain margin', the 'phase margin' and the 'crossover frequency'.

2. GENERAL REQUIREMENTS FOR THE DESIGN OF A CONTROL LOOP

Figure 1 shows the block diagram of the attitude control loop. In this loop the pilot acts as a controller by applying forces on the control stick and observing the pitch attitude.

This loop, closed via the pilot, can be considered as a general control loop. The classical requirements of control theory are valid. They include the following aspects:

1. The control loop has to be stable
2. A certain steady state accuracy must be attained
3. A sufficient damping is necessary
4. An adequate time response is needed

The 1st requirement is a necessary condition and the 2nd is fullfiled by including an integration term in the pitch attitude/control force transfer function. The remaining requirements may be evaluated by considering the frequency response characteristic of the open loop.

However, the characterization of this open control loop is very difficult because it includes the pilot. The pilot of course is a very complicated system and cannot be exactly described by a linear transfer function. Normally the pilot changes his gain, varies his time constants and shows a nonlinear behaviour. Therefore only simple pilot models are used, but their application may lead to an incorrect evaluation. For this reason here the pilot model is excluded by setting the simulated pilot to unity. Thus making the ratio pitch attitude/control force to be the open loop transfer function of the control loop.

Now the problem of control loop design is a conflict of the above 3rd and 4th requirement. If the damping is decreased the response time is less and vice versa. Therefore the design of a control loop always has to be a compromise between quality of damping and response time. Considering a pilot and a control engineer both should arrive at approximate-

ly the same solution. A control engineer would use a deterministic approach applying control system theory to design a controller. In case of the attitude control loop, the pilot takes over the function of the controller. He adapts himself intuitively in such a manner that an acceptable compromise between quality of damping and response time results. Therefore, the pilot evaluates intuitively the same parameters as the control engineer. This is the reason why we must look to the design parameters used in control theory.

3. GAIN AND PHASE MARGIN

According to control theory there are three parameters giving a good measure of quality of damping and response time. These are the 'crossover frequency', the 'gain margin' and the 'phase margin' of the open loop. These parameters are illustrated in figure 2 and defined as follows:

- Crossover frequency ω_c is the frequency at which the gain curve crosses the OdB-line.
- Gain margin G_m is the difference of gain between the OdB-line and the gain at the frequency with a phase lag of 180°.
- Phase margin ρ_m is the difference of phase between the phase at the crossover frequency and a phase lag of 180°.

The phase margin can be considered as an appropriate measure of closed loop damping, if the gain attenuates near the crossover frequency. In order to evaluate this attenuation the gain margin is needed as a further parameter. If the gain margin is low the gain attenuation is small and vice versa. Therefore we can say: The magnitudes of gain- and phase margin of the open loop express the quality of damping of the closed loop.

The crossover frequency ω_c is a measure of the response time. The larger the value of ω_c the shorter is the response time of the closed loop and vice versa. However, increasing the crossover frequency in order to shorten the response time generally lowers the gain- and phase margin and results in poor damping. Therefore the design of a control loop is a compromise between crossover frequency, gain and phase margin.

Now the crossover frequency is a function of the steady-state gain of pitch rate/control force which is proportional to normal load factor/control force. In practise the pilot controls with a larger gain than the assumed unity gain and thus shifts the crossover frequency. In attemping to get a good closed loop characteristic the pilot has to intuitively adjust his gain. If he selects a large gain in order to decrease the response time he reduces the gain- and phase margin which represents the system damping. Reducing the gain increases the response time, but improves the damping. By means of gain adjustment the pilot is able to find a compromise between damping and response time. Thus possible shifting of the crossover frequency and therefore the resulting response time is a function of the gain- and the phase margin.

Due to this we can say that the two parameters, gain and phase margin of the open loop frequency response characteristic, pitch attitude/control force, may be a good measure of both damping and response time.

4. DEVELOPMENT OF THE LIMITS

For development and validation of a criterion, based on gain and phase margin, the frequency response characteristics and the pilot comments from Neal and Smith (2) were used. The dimension of the gain curve is [deg/lb]. This investigation shows that envelopes of gain- and phase margin can be assigned to levels of handling qualities. Figure 3 presents these results in a gain/phase margin diagram.

Handling qualities of Level 1 assume a phase margin about 95 degrees and a gain margin above 20.5 dB. An increasing or decreasing of the phase margin needs a higher value of the gain margin to maintain Level 1. When decreasing either or both margins, the handling qualities degrade below Level 2 to Level 3. However, the most direct way to degrade handling qualities is to decrease both margins simultaneously.

In case of a phase margin less than 100° the boundaries between Level 1 and Level 2 and also between Level 2 and Level 3 indicate that a decreasing of the phase margin must be followed by an increasing of the gain margin to maintain the handling quality level. The reason is that lower values of phase margin correspond to a larger phase lag of the system. This means the system itself responds slowly and the pilot may shorten the response time by using a larger gain. This requires a sufficient gain margin.

An increase in phase margin corresponds to a shortening of response time. In this case the pilot is satisfied with a lower gain margin because he only has to control with a lower pilot gain to reach a desirable response ime. For phase margins larger than 100° and handling qualities of Level 1, an increasing of phase margin requires a higher value of gain margin. In this case the system quickly responds and the pilot prefers a high damping for good handling qualities.

Up to now an upper limit of the Level 1 area cannot be proven until more flight testing is done in this area. It is assumed that a limit exists in the vicinity of a gain margin of 35 dB.

The maximum amount of the gain curve deviation from the 20 dB/decade line ($|\ |_{Max}$, figure 2) proves to be a further parameter influencing the handling qualities. For Level 1 it must be less than 10 dB and for Level 2 less than 20 dB.

Figure 4 shows typical pilot comments for the different areas in the gain/phase margin diagram. If we keep a constant gain margin it has been found that with an increasing of phase margin, the comments change from 'slow' to 'good' to 'too fast' initial response. If we keep a constant phase margin a decreasing of gain margin results in comments varying from 'steady on target' (Level 1) to 'tendency to oscillation' (Level 2) to 'PIO-tendency (Level 3). Obviously the value of the gain margin, depending on the phase margin, is a measure of PIO-tendency.

5. COMPARISON WITH MIL-F-8785

The so called gain/phase margin criterion is able to evaluate low and high order systems. For low order systems the evaluation should yield the same as given in the MIL-specifications. Therefore the criterion has been further proven by evaluating low order systems. The selected transfer function is presented by the following equation:

$$\frac{\Theta}{F_s} = \frac{K}{s} \cdot \frac{1}{1+Ts} \cdot \frac{1 + T_\Theta s}{1 + 2\zeta/\omega_n s + 1/\omega_n{}^2\ s^2} \tag{1}$$

Hereby gain K is the steady-state ratio of pitch rate/control force which is proportional to the ratio normal load factor/control force (equation (2)).

$$K = \left(\frac{\dot{\Theta}}{F_s}\right)_{\text{Steady-State}} = \left(\frac{n}{F_s}\right)_{\text{Steady-State}} \cdot \frac{g}{V_{TAS}} \tag{2}$$

The limits of the ratio control force/normal load factor (F_s/n) are defined in Para. 3.2.2.2.1 of MIL-F-8785 (1). For further calculation it has been assumed that F_s/n has a value of 5 lb/g. With a selected true airspeed V_{TAS} of 800 ft/sec a value of $K = 0.46$ deg/sec lb results. The term $1/(1+Ts)$ of Eq. (1) represents the control system. For the lag time constant T a value of 0.05 sec has been selected. According to Para. 3.5.3 of MIL-F-8785 B this value causes the allowed phase lag of 30° at $\omega = 11.5$ rad/sec and therefore fullfils the requirements of Level 1 up to this frequency. The lead time constant T_Θ has been varied from 0 to 2 sec. The damping ratio ζ and the natural frequency ω_n have been varied within the limits of the MIL-specification.

Figure 5 presents the result of a lead time variation. F_s/n and ω_n has been kept constant within optimum of Level 1 ($n/\alpha = 20$ g/rad). With a damping ratio of $\zeta = 0.3$, which is near the lowest allowed limit, even high value of the lead time constant do not bring the system to Level 1. An increasing of the damping ratio to an acceptable value of $\zeta = 0.67$ results in Level 1 in the range of lead time constants from 0.2 sec to 0.8 sec. A further increasing of the damping ratio ($\zeta = 1.5$) produces a slower system and therefore higher values of the lead time constant are necessary for Level 1 (~ 0.4 sec $< T_\Theta < \sim 1.5$ sec). The optimal area of Level 1 is reached with optimal values of damping ratio, natural frequency and proper values of normal load factor/control force and lead time constant. The gain/phase margin criterion is in agreement with the MIL-specifications.

Figure 6 indicates the effect of variation of the natural frequency. F_s/n and ζ are in an optimum range of the MIL-specifications. Without any lead only high values of ω_n ($\omega_n > 10$ rad/sec) produce Level 1 handling qualities, and then only within a limited area. However, with $\omega_n = 30$ rad/sec handling qualities must be degraded to Level 3 because the maximum amount of the gain curve ($|\ |_{MAX}$, figure 2) is larger than 20 dB. A proper lead time of 0.5 sec brings the system to Level 1 for a range of ω_n from approximately 2.5 rad/sec to 7 rad/sec. The configuration with $\omega_n = 30$ rad/sec is again degraded to Level 3 ($|\ |_{MAX} > 20$ dB). A lead time of 1 sec increases the initial response and therefore the frequency range of Level 1 decreases (from 2 rad/sec to about 4 rad/sec).

Figure 7 presents the result of a variation of the damping ratio ($\zeta = 0.3 \div 1.5$). In case without lead Level 1 area cannot be reached. With a lead time constant of 0.5 sec the criterion gives the same evaluation as the MIL-specifications. A further increase to $T_\Theta = 1$ sec, which provides a faster initial response, needs a higher value of the damping ratio. This evaluation indicates that with a proper value of T_Θ the MIL-limits of the damping ratio ζ are correct, but that with an increase of the lead time constant the limits must be reduced accordingly.

Figure 8 shows the effect of a variation of the steady-state value of F_s/n. This factor influences the gain/phase margin and therefore the handling qualities. Without any lead only a higher value of F_s/n causes a Level 1 rating. Again with $T_\Theta = 0.5$ sec a confirmity with the MIL exists. In case of larger lead time constants higher values of F_s/n are necessary for Level 1.

Figure 9 indicates the effect of an additional time delay. The parameters of the transfer function are all inside of the Level 1 limits of the MIL-specifications. Independent of the lead time constants all systems with $\tau = 0$ are within the Level 1 area. Already a small time delay degrades the handling qualities from Level 2 to Level 3. A time delay of only 0.05 sec causes a degradation from the optimal area to the boundary of Level 1/Level 2.

6. ADVANTAGES

Due to the recognizable effect of various parameter variations an advantage of this criterion emerges: it facilitates the design of good handling qualities. For example figure 10 shows clearly how a case of bad handling qualities, caused by a time delay, is improved by a higher value of F_s/n. Figure 10 also shows how a decreasing damping ratio may be improved by increasing the ratio of control force/normal load factor and vice versa.

A further advantage of the gain/phase margin criterion is a possible PIO-evaluation. As presented in figure 4, PIO-tendency increases by a decrease of the gain margin. Figure 11, which includes the effect of all parameter variations, indicates that the variation of ζ, F_s/n and τ directly shifts the gain margin. These parameters are mainly responsible for PIO-tendencies. An increasing of the time delay, a decreasing of the damping ratio as well as a decreasing of the ratio control force/normal load factor causes an increasing PIO-tendency. The variation of the remaining parameters T_Θ and ω_n have a lower influence on PIO-tendencies because their direction of shift is almost parallel to the phase margin line. These results are confirmed by special PIO-criteria as (3) and (4).

7. DESIGN OF A FLIGHT CONTROL SYSTEM

The gain/phase margin criterion has already been applied to the design of a Command and Stability Augmentation System (CSAS) for a high performance fighter in a Delta-Canard-configuration. The object has been to calculate the coefficients of the given flight control system in such a way that handling qualities of Level 1 result according to both the MIL-specifications and the gain/phase margin criterion. The design has been made over the whole flight envelope, which requires a large number of flight cases. For a true airspeed V_{TAS} less than 800 ft/sec, the steady-state ratio of pitch rate/control force has been selected to 0.46 deg/sec lb while for a true airspeed above this value, the ratio of pitch rate/control force has been calculated according to Eq. (2) with $F_s/n = 5$ lb/g. Applying the gain/phase margin criterion the frequency response characteristic of pitch attitude/control force has been used whereas for the MIL-requirements the roots of the airframe plus 'CSAS' has been chosen. These roots represent the equivalent short-period characteristic.

Figure 12 gives the handling qualities evaluation of these two criteria. The damping ratios and the natural frequencies of the equivalent short period are within the limits of Level 1. The lead time constant ranges from about 2 sec to about 0.5 sec. This indicates that this time constant is situated in the area of high performance fighter. At the same time all design points are also within the limits of Level 1 of the gain/phase margin-criterion, even near the optimal area. As a result it has been found that there are no obvious discrepancies between these two criteria. However, it must be mentioned that the evaluation according to MIL-specifications only considers the roots representing the short-period chararacteristic and neglects the further roots of the transfer function of the closed control loop.

This design of a flight control system has been accomplished to fullfil both of the criteria. This indicates that the gain/phase margin-criterion may yield to the same results as the MIL-specifications.

8. RULES OF APPLICATION

The gain/phase margin-criterion needs the frequency response characteristic of pitch attitude/stick force including the steady-state ratio. This characteristic shall not include the dynamic characteristic of the cockpit controller. The considered frequencies range from about 0.4 rad/sec to about 10 rad/sec.

If only the frequency response characteristic of the pitch rate/control force is available that of pitch attitude/control force can be calculated by the addition of a phase lag of 90° and the addition of a gain line with an attenuation of 20 dB/decade. Thereby the crossover frequency of this line is determined by the steady-state value of pitch rate/control force (figure 2).

In case the steady-state value of pitch rate/control force is not known, this parameter can be calculated according to Eq. (2) by using the values of control force/normal load factor and the value of true airspeed. If the steady-state ratio cannot be calculated because the value of F_s/n is not known, it may be assumed that for V_{TAS} larger than 800 ft/sec, F_s/n has a value of 5 lb/g and for V_{TAS} less than 800 ft/sec, the steady-state ratio of $\dot{\theta}/F_s$ has a value of 0.46 deg/sec lb.

When the value of F_s/n is within the limits of para. 3.2.2.1 of the MIL-F-8785 C, the the handling qualities of the longitudinal maneuvering characteristics can be evaluated as follows:

1. Calculation of the gain margin G_m (figure 2)
2. Calculation of the phase margin φ_m (figure 2)
3. Calculation of the maximal gain amount $| \ |_{MAX}$ (figure 2)
4. Evaluation according to the gain/phase margin diagram (figure 3)
5. Degradation to Level 2 if $| \ |_{MAX} > 10$ dB and to Level 3 if $| \ |_{MAX} > 20$ dB

9. CONCLUSION

The parameters gain and phase margin of pitch attitude/control force has proven to be a reliable measure of the longitudinal maneuvering characteristics with and without a Command and Stability Augmentation System. In the same manner as the MIL-specifications the gain/phase margin criterion defines ranges for different levels of handling qualities. The application of the described method is very simple to use and does not require extensive calculations.

The criterion gives an overall evaluation of all systems situated between the control force and the pitch attitude. Compared with the MIL-specifications, this method offers the advantage of a complete evaluation of all parameters such as damping ratio and natural frequency of the short period, lead and lag time constants, time delay and even the value of control force/normal load factor. This means the criterion evaluates the dynamic characteristics as well as the steady-state. This makes it possible to perform a PIO-evaluation.

This criterion does not overcome the general weakness of neglecting nonlinearities. Further it does not include the dynamic behaviour of the control stick and the ratio of control force to stick deflection. There is another aspect which is generally neglected but must be considered in future: the speed of pilot adaption capability.

Our effort in the future will be a further improvement of this criterion so that it will be applicable to different classes of airplanes and different categories of flight tasks. Special attention will be given to PIO-evaluation.

10. REFERENCES

[1] Anon, Military Specification, Flying Qualities of Piloted Airplanes, MIL-F-8785 C

[2] Neal, T.P., Smith, R.E.: An In-Flight Investigation to Develop Control System Design Criteria for Fighter Airplanes (USAF) AFFDL-TR-70-14, Volumes I and II, December 1970

[3] A'Harrah, R.C., Siewert, R.F.: Pilot Induced Instability, AGARD Flight Mechanics Panel, Specialist Meeting on Stability and Control, Cambridge, England, September 1966

[4] DiFranco, D.A.: Flight Investigation of Longitudinal Short Period Frequency Response and PIO Tendencies, AFFDS-TR-66-163, 1967

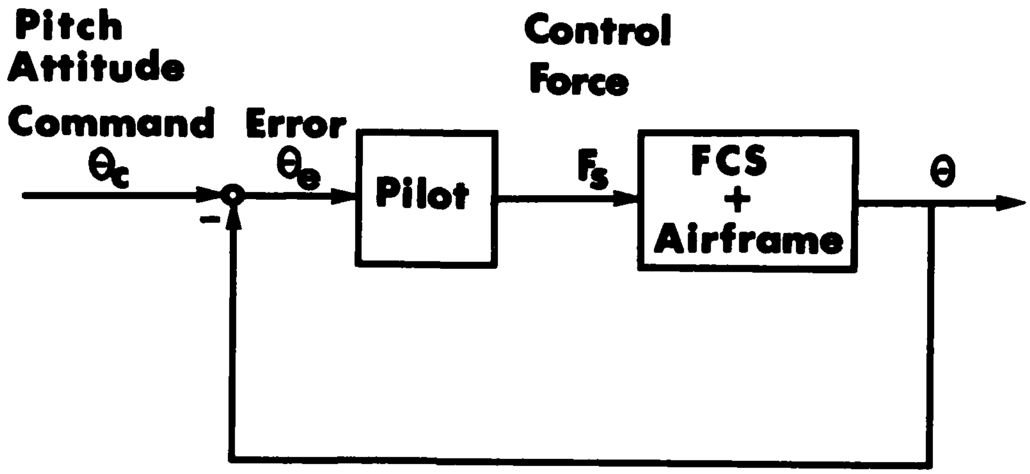


FIGURE 1: BLOCK DIAGRAM OF THE ATTITUDE CONTROL LOOP

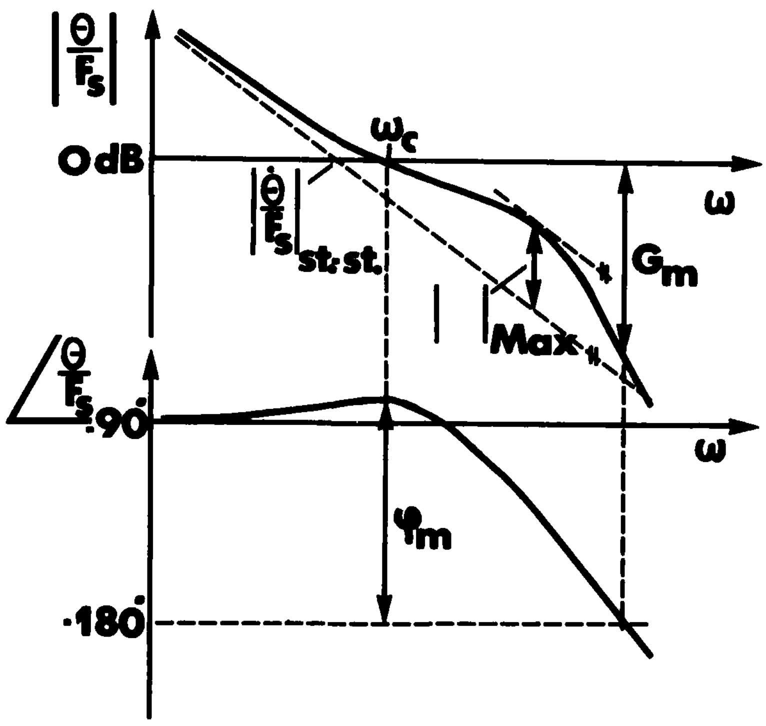


FIGURE 2: OPEN LOOP FREQUENCY RESPONSE CHARACTERISTIC

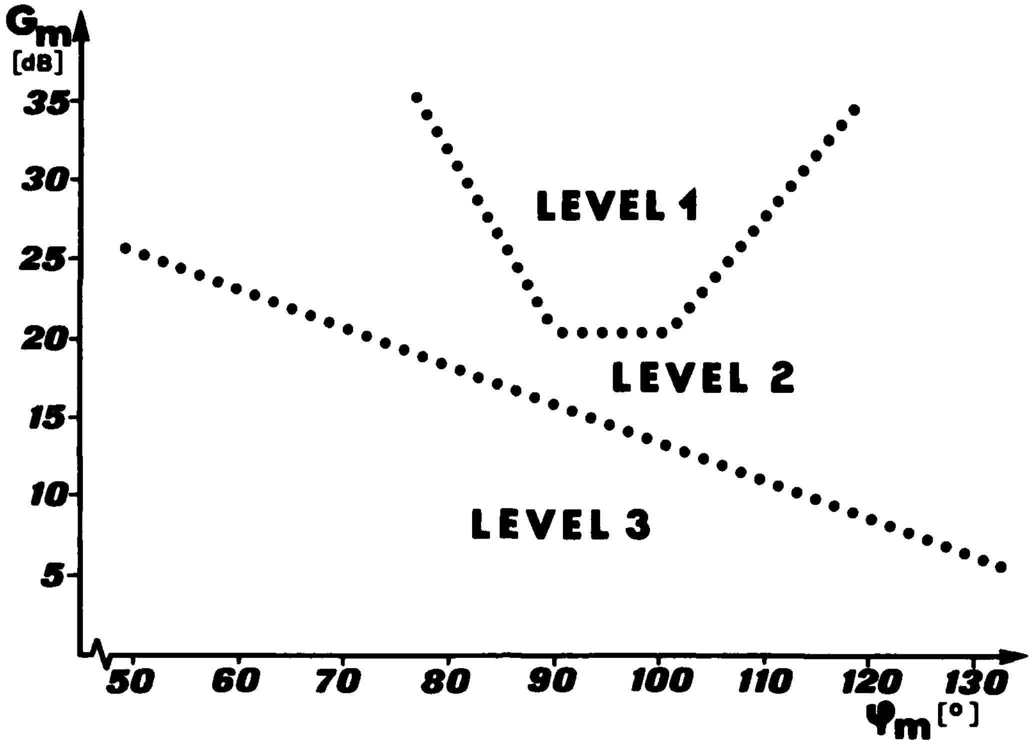


FIGURE 3: GAIN- AND PHASE MARGIN CRITERION

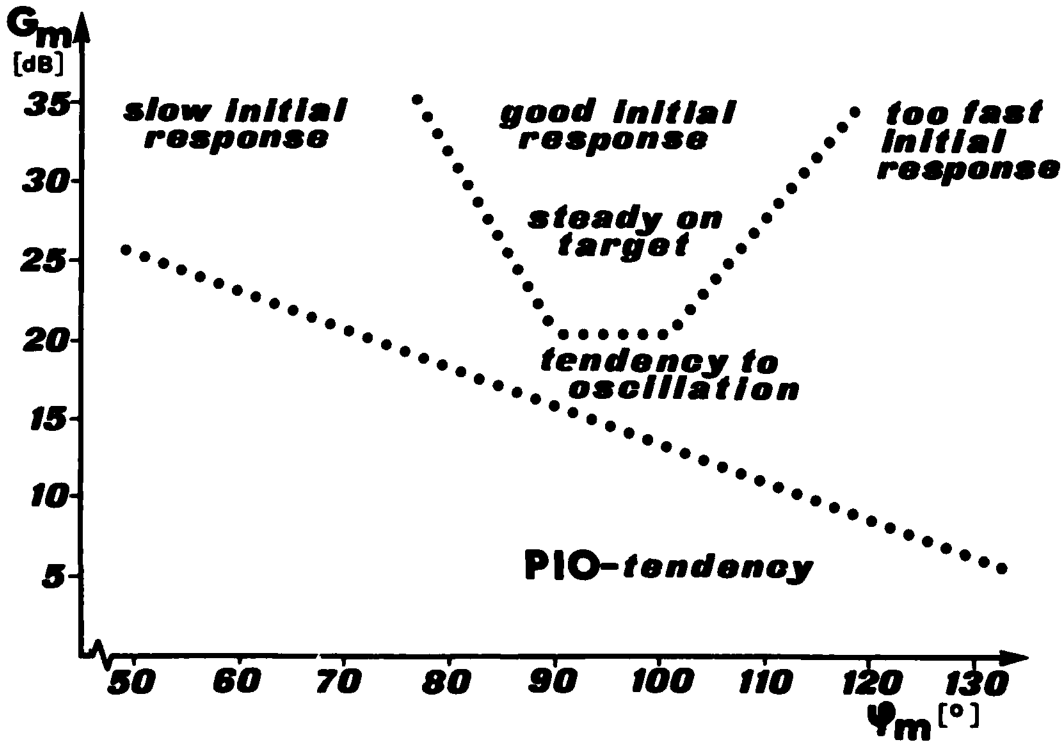


FIGURE 4: PILOTS' COMMENTS

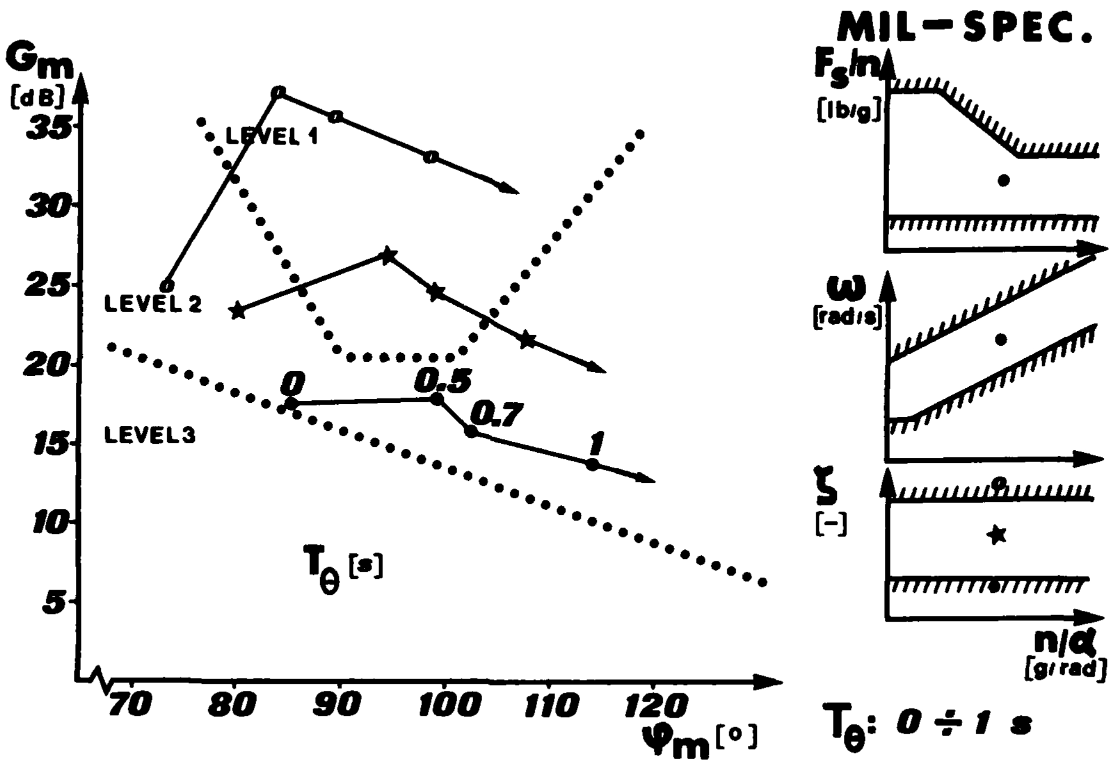


FIGURE 5: EFFECTS OF T_θ-VARIATION

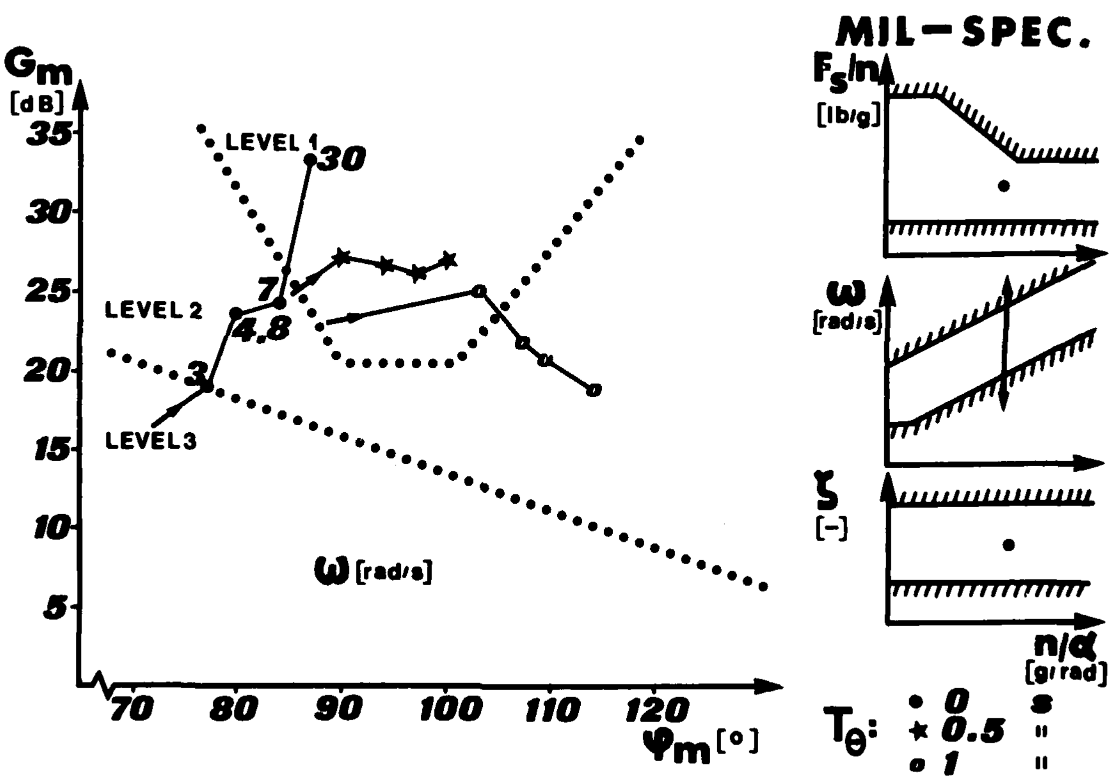


FIGURE 6: EFFECTS OF ω_n-VARIATION

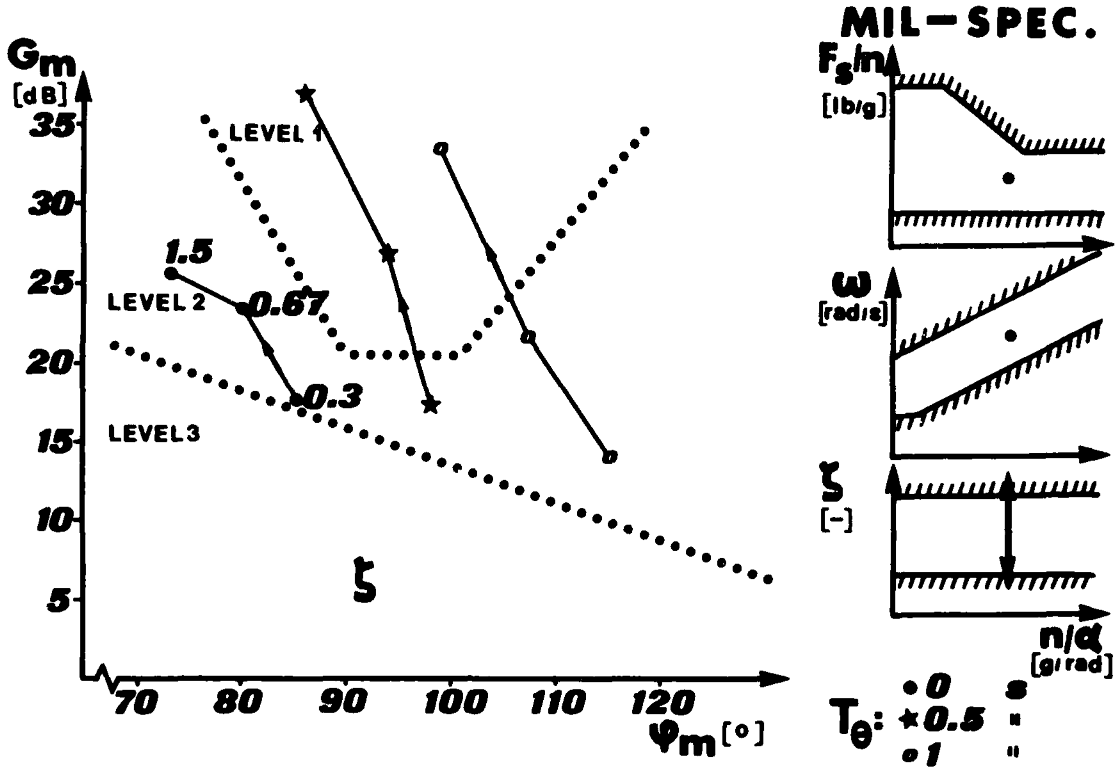


FIGURE 7: EFFECTS OF ζ-VARIATION

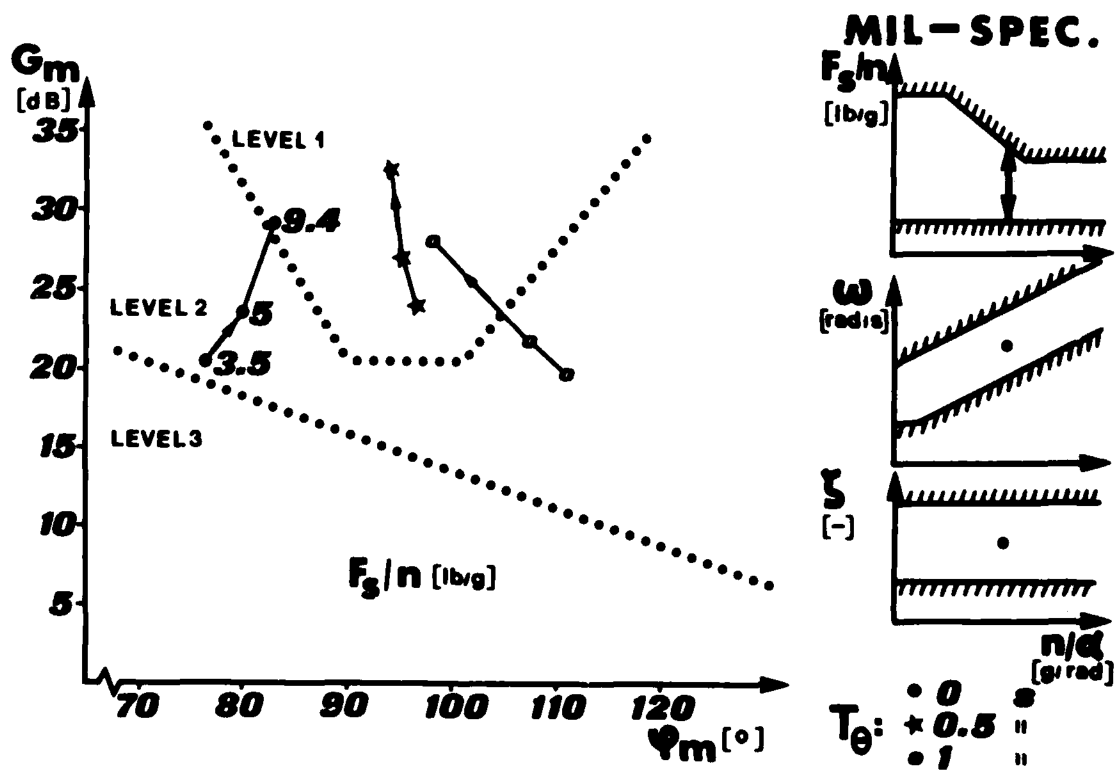


FIGURE 8: EFFECTS OF F_s/n-VARIATION

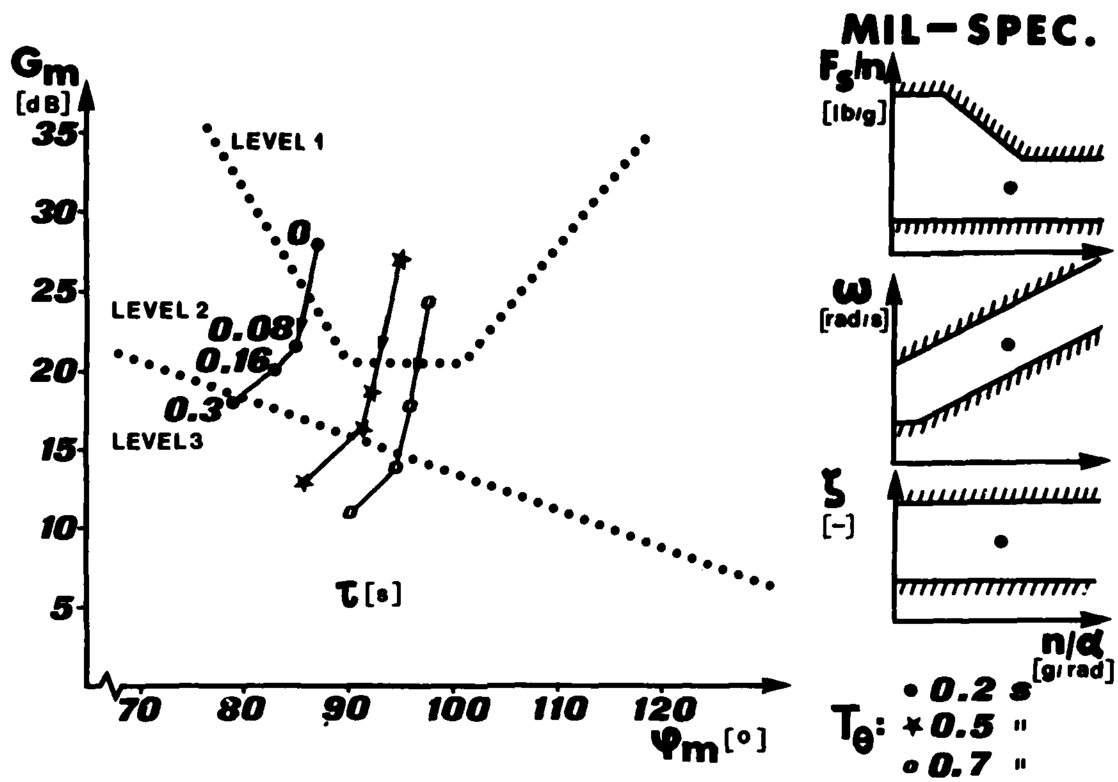


FIGURE 9: EFFECTS OF AN ADDITIONAL TIME DELAY

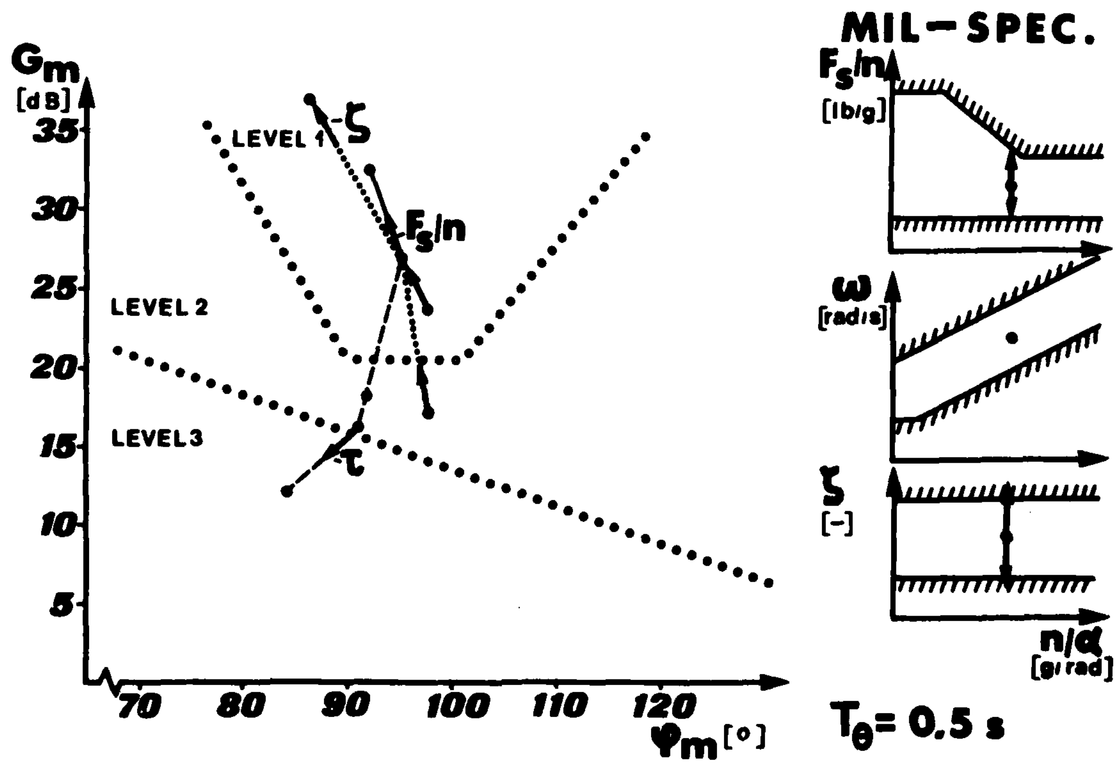


FIGURE 10: COMPENSATING ζ AND τ BY F_s/n

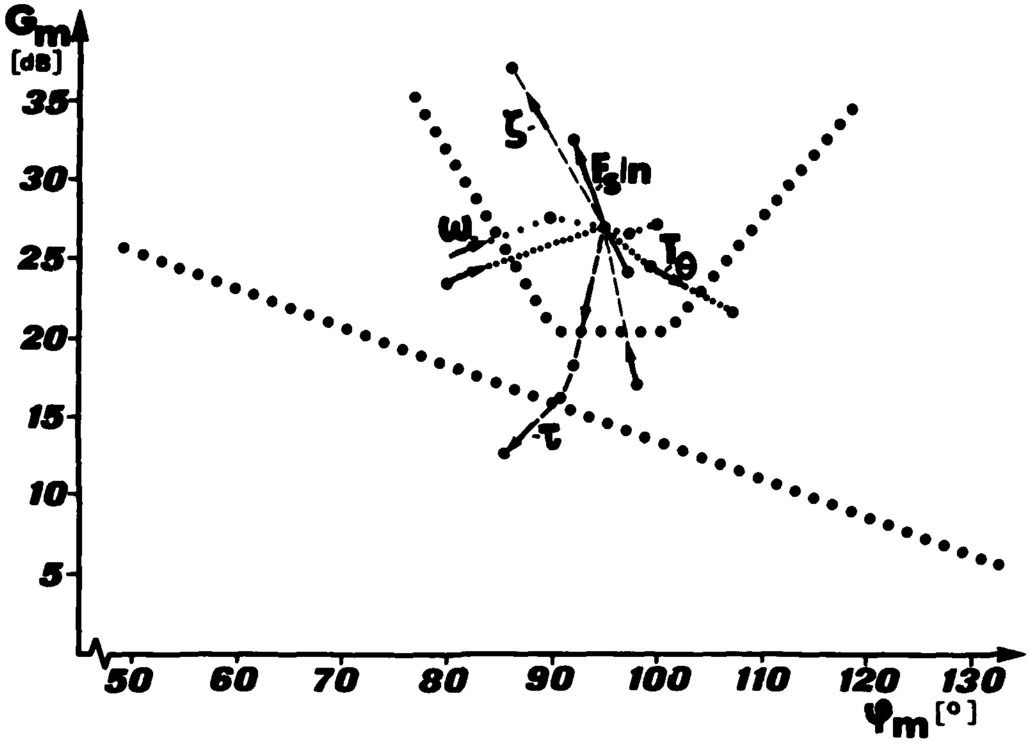


FIGURE 11: PIO-TENDENCIES OF VARIOUS PARAMETERS

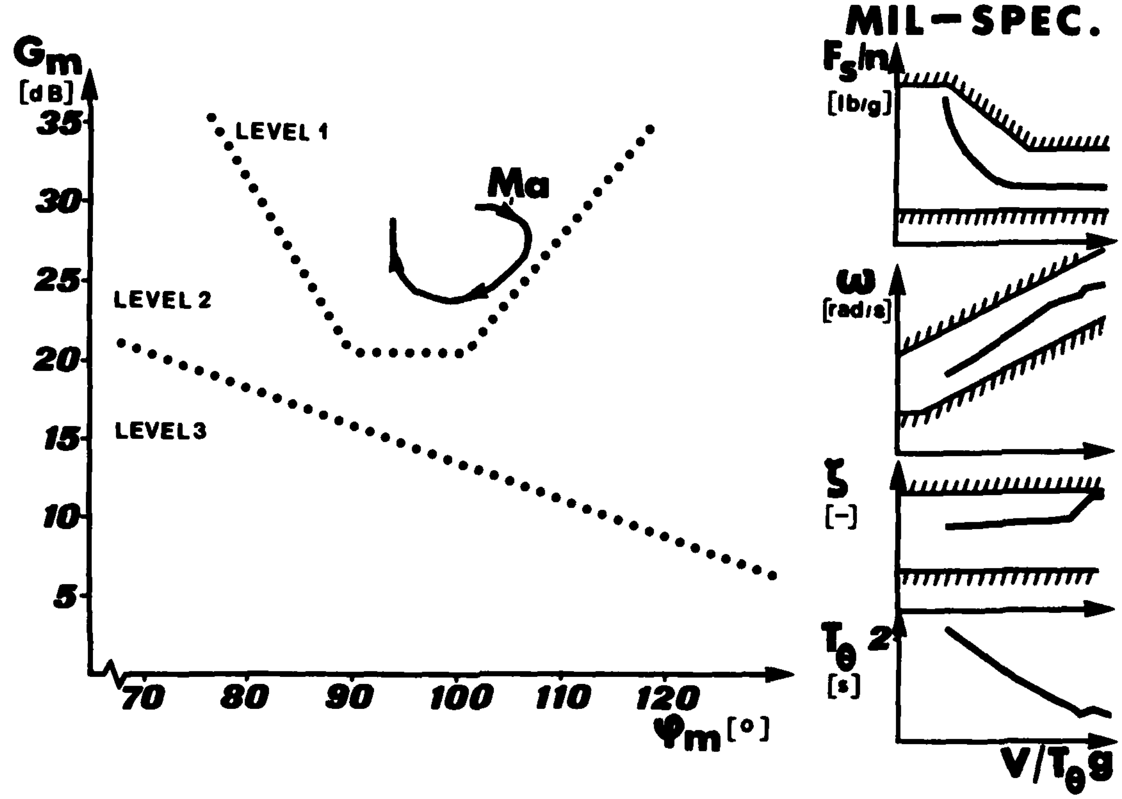


FIGURE 12: COMPARISON GAIN-PHASE MARGIN CRITERION/MIL-F-8785

LES COMMANDES DE VOL ELECTRIQUES :

VERS DE NOUVELLES NORMES DE JUGEMENT DES QUALITES DE VOL

UN EXEMPLE : LE MIRAGE 2000

par

Paul-Louis MATHE

Chef du Département Dynamique du Vol
AVIONS MARCEL DASSAULT-BREGUET AVIATION
78, quai Carnot - 92214 ST CLOUD (France)

Le MIRAGE 2000 est un avion équipé d'un système de commandes de vol strictement électriques. Le choix d'une telle solution répondait à deux objectifs principaux.

Le premier de ces objectifs était de rendre l'avion capable de voler dans des conditions d'instabilité longitudinale naturelle notable. Les avantages directs de telles conditions de vol sont bien connus et se traduisent en termes de gains de performances, en particulier sur les vitesses d'approche, sur les marges et limites de manoeuvres, tant subsoniques que supersoniques. Ces gains sont particulièrement sensibles pour un avion de formule Delta comme le MIRAGE 2000. Les avantages indirects ne sont pas moins importants : la suppression de la contrainte de stabilité longitudinale dans l'optimisation aérodynamique de l'avion a permis de retenir des dispositifs tels que des becs de bord d'attaque de grandes surfaces, dont les effets sur la finesse sont spectaculaires, mais dont les conséquences sur la stabilité longitudinale sont telles que leur adoption pour un avion équipé de commandes de vol classiques serait délicate. Enfin, l'adoption d'un système de commandes de vol électriques ayant la capacité d'assurer la stabilisation d'un avion naturellement instable, permet de réduire à de simples problèmes de compatibilité structurale les exigences d'emport de multiples charges externes sans conséquence vraiment pénalisante sur la définition générale de l'avion.

Ce premier objectif, visé par l'adoption de commandes de vol strictement électriques sur le MIRAGE 2000, est fondamental. La conception même de l'avion en dépendait. C'est lui qui permet d'exploiter pleinement les avantages de la formule Delta (avantages structuraux et aérodynamiques) sans en subir le principal handicap qui réside dans l'impossibilité de l'équiper de dispositifs hypersustentateurs efficaces.

Le second de ces objectifs était de réaliser, par rapport à un avion équipé de commandes de vol classiques, un pas significatif dans l'amélioration des qualités de pilotabilité de l'avion. Ce second objectif, bien que n'étant pas en soit fondamental pour la conception de l'avion, n'en est pas moins primordial dans le bilan final exprimé en termes d'efficacité opérationnelle.

o
o o

Le premier des objectifs que nous avons décrit a exigé, pour être atteint, un effort de conception technologique du système de commandes de vol électriques, afin d'assurer un niveau suffisamment élevé de sécurité. Deux grands principes ont présidé à cette conception :

- choix d'un niveau de redondance adapté à la criticité des fonctions réalisées par les différents éléments du système ; en particulier, une redondance d'ordre 4 (capacité de survie à une double panne) a ainsi été retenue pour l'élaboration des ordres de contrôle des élevons et seulement d'ordre 3 pour l'élaboration des ordres de direction.

- capacité d'une certaine reconfiguration fonctionnelle des chaînes en fonction du niveau d'intégrité de ses constituants.

Le deuxième objectif n'ajoutait en soit que peu de difficultés techniques à celles qu'il fallait surmonter pour atteindre le premier objectif : un système de commandes de vol électriques, capable d'assurer la stabilisation d'un avion naturellement instable, possède la variété de détecteurs et la capacité de traitement exigée par une optimisation poussée du comportement général de l'avion vu du pilote, c'est-à-dire de sa pilotabilité. En fait, les difficultés résidaient a priori beaucoup plus, d'une part dans le choix même des objectifs de pilotabilité à viser, d'autre part dans celui des critères à prendre en compte pour l'établissement des réglages du système de commandes de vol permettant d'atteindre ces objectifs.

Le prototype n° 1 du MIRAGE 2000 effectua son premier vol, le 10 Mars 1978. Son système de commandes de vol électriques avait alors atteint un stade de développement suffisamment avancé, et la confiance que l'on pouvait avoir en son bon fonctionnement, assise sur de multiples campagnes d'essais sur banc au sol, était suffisante pour que les responsables des essais en vol de l'avion n'hésitent pas à lui faire ouvrir, dès ce premier vol, un domaine incluant des vitesses supersoniques (Mach 1.3).

Ce premier vol et les suivants confirmèrent la validité des solutions technologiques retenues et celles des principaux réglages des commandes de vol qui régissent le comportement statique et dynamique de l'avion piloté, c'est-à-dire en définitif ses qualités de vol.

Et il est important de préciser que les principaux réglages, qui ont volé au cours des premiers vols du premier prototype, ont à quelques détails près été conservés pour les commandes de vol du MIRAGE 2000 de série. Certes quelques ajustements ont été indispensables au cours des essais en vol. Mais ces ajustements furent minimes. Par ailleurs, s'ils se sont avérés nécessaires, c'est qu'en raison de certaines divergences découvertes en vol, essentiellement en transsonique, entre les véritables caractéristiques aérodynamiques de l'avion et nos prévisions, le comportement réel de celui-ci s'est révélé différent du comportement souhaité, et non parce que le comportement souhaité ne satisfaisait pas les pilotes. Ceci signifie bien que les critères de qualités de vol, pris en compte pour l'établissement, avant vol, des règlages du système de commandes de vol, ont été parfaitement vérifiés par l'expérimentation.

C'est un succès certain ; mais, en toute honnêteté, il n'y a pas lieu d'en tirer une gloire excessive, car ce succès est parfaitement logique et la conséquence directe des possibilités offertes par les commandes de vol électriques.

En effet, tous les critères de qualités de vol, qu'il s'agisse de critères universellement connus (au point qu'ils constituent la base de certaines normes) ou de critères moins connus et propres au constructeur, présentent la particularité de n'être significatifs, et donc vraiment applicables, que sous deux conditions :

- qu'ils concernent des comportements de l'avion relativement "simples" : modes découplés, quasi-linéarité de ces modes, ...
- qu'ils soient vérifiés très "largement", c'est-à-dire que l'on reste dans leur application toujours très éloigné des limites de validité de ces critères.

Or les possibilités offertes par les commandes de vol électriques permettent aisément de remplir ces deux conditions ; elles permettent de "simplifier" le comportement de l'avion vu du pilote, en masquant les éventuels accidents aérodynamiques d'une part, en découplant les modes de comportement de l'avion sur ses trois axes d'autre part. Elles permettent, par ailleurs, de réaliser une très large gamme de réglage des paramètres principaux de comportement de l'avion, et donc de satisfaire ces critères au plus proche de leur optimum.

Il n'est donc pas étonnant, dans ces conditions, qu'aucune difficulté n'ait été rencontrée en vol pour vérifier la validité des critères utilisés pour l'établissement des règlages du système des commandes de vol du MIRAGE 2000.

Encore fallait-il, pour assurer ce succès, que deux conditions soient remplies :

- un choix judicieux des critères,
- une réalisation matérielle des divers éléments constitutifs du système de commandes de vol (électronique d'élaboration des ordres, servocommandes électro-hydraulique de puissance) à la hauteur des ambitions de ces critères.

La très longue expérience acquise par les AVIONS MARCEL DASSAULT - BREGUET AVIATION, qui rappelons-le, ont toujours conçu et produit eux-mêmes les éléments essentiels des systèmes de commandes de vol de ses propres avions, a bien aidé à remplir ces deux conditions. Cette expérience s'appuie sur une suite continue de réalisations (MIRAGES III, MIRAGES à décollage vertical, MIRAGES F, MIRAGES G à géométrie variable, etc...) dont les systèmes de commandes de vol préfiguraient, sous de multiples aspects, le système de commandes de vol strictement électriques du MIRAGE 2000 et dont l'expérimentation en vol a permis de dégager un certain nombre de "règles de l'art" en matière de qualités de pilotabilité.

o
o o

Cependant, un examen détaillé du schéma fonctionnel des commandes de vol du MIRAGE 2000 de série, comparé au schéma de ce qui volait en 1978 sur le prototype, montre des différences notables. Ce fait n'est pas en contradiction avec ce que nous venons d'exposer ; il ne traduit pas une mise en cause des critères de comportement à vérifier pour satisfaire certains objectifs de pilotabilité, mais une évolution de ces objectifs, souhaitée par les pilotes, au fur et à mesure de leur découverte des possibilités qu'offraient les commandes de vol électriques. Et il est bien certain que les objectifs atteints par les commandes de vol du MIRAGE 2000 de série dépassent très largement ceux que nous nous étions fixés préalablement à tout essai en vol et que nous avions atteints avec le premier prototype du MIRAGE 2000.

Cette évolution des objectifs de pilotabilité est très caractéristique de la situation nouvelle créée par l'apparition des commandes de vol électriques : pleinement satisfaits des qualités de pilotabilité (stabilité de la plateforme, temps de réponse aux ordres de commande, "pureté" de comportement,....) que fournissent de tels systèmes de commandes de vol, les pilotes ont progressivement déplacé leurs exigences dans le sens d'une simplification, voire d'une suppression complète, des consignes de pilotage exigées par le respect du domaine d'emploi de l'avion. Et c'est ainsi qu'a été développé, dans les commandes de vol du MIRAGE 2000, un certain nombre de fonctions de limitations automatiques de plus en plus perfectionnées.

Cette évolution peut être illustrée par deux exemples vécus au cours de la mise au point des commandes de vol du MIRAGE 2000.

Le premier de ces exemples concerne le contrôle longitudinal de l'avion et ses limitations en incidence et en facteur de charge.

Dans le choix des fonctions et des réglages de la chaîne de tangage du MIRAGE 2000, un certain nombre d'options avait été initialement retenu et c'est d'ailleurs dans la configuration correspondante que les premiers vols furent effectués. Parmi ces options, les deux principales étaient les suivantes :

- Le système de commandes de vol comporterait un dispositif assurant une limitation automatique des incidences de l'avion de façon à supprimer les risques de perte de contrôle en manoeuvres. Il avait alors été admis qu'un tel dispositif serait opérationnellement très intéressant, même si son efficacité était limitée aux configurations de combat (masse et configuration de charges externes) les plus courantes et pour des vitesses supérieures à une valeur de l'ordre de 100 kts. Ceci supposait donc que pour certaines configurations de combat et en dessous de cette vitesse, la sécurité vis-à-vis des pertes de contrôle était assurée par des consignes de pilotage.
- Hors de la zone de domaine de vol dans laquelle l'avion est limité par son incidence maximale, le facteur de charge maximum qu'il serait possible d'obtenir, manche en butée, serait compris entre le facteur de charge limite et le facteur de charge extrême de l'avion, ce qui impliquait, évidemment, une consigne particulière de pilotage.

C'est dans cette configuration de commandes de vol que les premiers vols de l'avion furent effectués. Les pilotes furent très satisfaits de cette solution .... mais pas longtemps.

De nouvelles exigences, c'est-à-dire de nouveaux objectifs de pilotabilité, furent alors précisées. Le choix d'un réglage permettant d'obtenir, sur la butée du manche en profondeur, un facteur de charge supérieur au facteur de charge limite de l'avion fut très vite critiqué, bien que ce choix ait été retenu à la demande même des pilotes en vertu du principe de sécurité selon lequel il vaut mieux "plier" un avion que de percuter le sol. La critique portait essentiellement sur la difficulté à respecter scrupuleusement les facteurs de charge limites de l'avion : ou le pilotage du facteur de charge en manoeuvre était relativement "lâche" et, dans ces conditions, de nombreux dépassements du facteur de charge limite pouvaient être observés avec toutes les conséquences que ces dépassements peuvent avoir sur la fatigue de la structure et du pilote, ou le pilotage du facteur de charge, pour éviter ces dépassements, était un pilotage "serré" et, dans ces conditions, la charge de travail du pilote était augmentée ainsi d'ailleurs que les temps de réponse effectifs de l'ensemble avion + pilote. Pour satisfaire le principe de sécurité précédemment décrit et le souhait d'avoir une limitation automatique du facteur de charge de l'avion à la valeur limite de la structure, nous avons adopté un système de restitution d'efforts à double butée : une butée élastique sur laquelle le facteur limite est obtenu ; une butée mécanique permettant d'obtenir un facteur de charge compris entre le facteur de charge limite et le facteur de charge extrême de l'avion.

Pour un certain nombre de configurations de charges externes très lourdes, le facteur de charge limite de l'avion est réduit. Initialement, il avait été prévu que le respect de cette limitation serait réalisé par simple consigne de pilotage. Là encore, une évolution très nette des souhaits des pilotes se fit jour au fur et à mesure de l'expérimentation en vol de l'avion et aboutit à l'adoption d'une commutation, à la disposition du pilote, lui permettant de choisir, selon la configuration de son avion, la valeur du facteur de charge limite automatiquement respecté par le système des commandes de vol pour un ordre du pilote correspondant à la butée élastique du manche.

Le dispositif de limitation automatique de l'incidence, principale clé de la protection de l'avion vis-à-vis des pertes de contrôle, subit de même un certain nombre d'évolutions allant toutes dans le sens d'une diminution des contraintes de pilotage. Le premier souhait des pilotes fut que toutes les configurations de combat de l'avion soient couvertes par ce dispositif. Le second, que la protection vis-à-vis de la perte de contrôle, pour ces configurations, soit étendue jusqu'à la vitesse nulle. Ce qui fut fait mais au prix, on le conçoit, d'une complexité accrue du dispositif.

Le second exemple concerne, cette fois-ci, le contrôle transversal de l'avion. La protection vis-à-vis d'éventuelles pertes de contrôle dépend, nous l'avons vu, de la qualité du dispositif de limitation automatique d'incidence ; elle dépend aussi de la qualité du contrôle transversal de l'avion au cours de manoeuvres à grande incidence. Cette qualité est obtenue par une adaptation soignée des chaînes de roulis et de lacet du système de commandes vol. Cette adaptation, réalisée dans l'hypothèse d'un contrôle transversal de l'avion au gauchissement seul, peut être mis en défaut, dans certaines conditions particulières, si le pilote pour ces manoeuvres veut "s'aider" du pied. La consigne de pilotage, qui en résulte, de limiter les ordres au palonnier dans ce genre de manoeuvres, fut très rapidement critiquée par les pilotes et tout un jeu de modifications de la chaîne transversale fut appliqué pour qu'en aucun cas une manoeuvre intempestive du palonnier ne puisse mettre en défaut le contrôle transversal de l'avion à grande incidence.

Et c'est ainsi que le MIRAGE 2000 est devenu un avion pour lequel les consignes restrictives de pilotage ont pratiquement complètement disparu : quelles que soient les manoeuvres du pilote sur ses commandes, la protection de la structure, ainsi que la protection de l'avion vis-à-vis des pertes de contrôle sont automatiquement assurées et ce, jusqu'à vitesse nulle.

o
o o

Nous avons cité ces quelques exemples, vécus au cours de la mise au point du système de commandes de vol du MIRAGE 2000, pour souligner l'évolution des normes de jugement de qualités de vol qui semblent aujourd'hui se dessiner avec l'apparition d'avions équipés de systèmes de commandes de vol électriques.

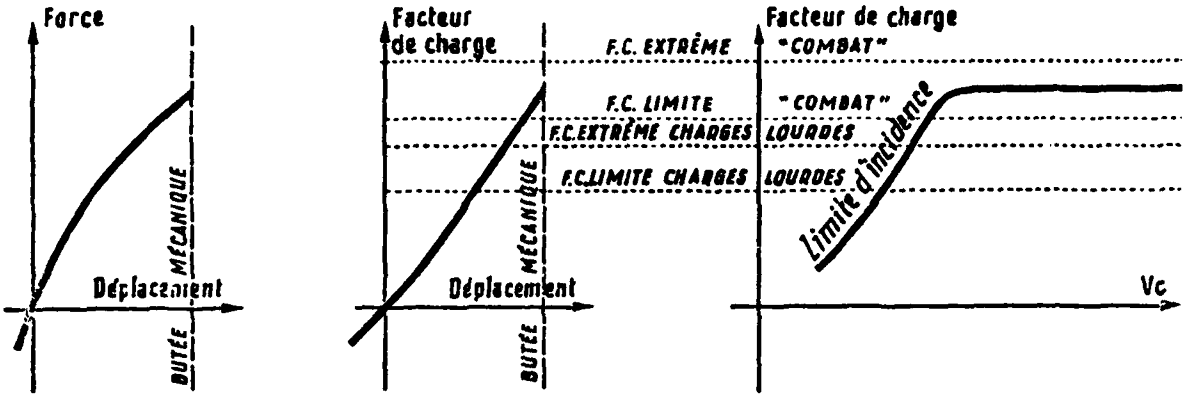
De tels systèmes de commandes de vol permettent, sans trop de difficulté, de "modeler" le comportement piloté de l'avion de façon très souple et très efficace et de réaliser ainsi une optimisation, aussi poussée qu'on le souhaite, des qualités de vol de l'avion au sens classique du terme. C'est ce résultat qui conduit les pilotes à déplacer leurs exigences : pleinement satisfaits des qualités de vol que le système de commandes de vol électriques confère à l'avion dans son domaine normal d'utilisation, ils souhaitent que ce système les aident à respecter les limites de ce domaine. Ce souhait est d'ailleurs d'autant plus justifié que, au moins dans un certain nombre de cas, la pureté de comportement de l'avion, assurée par les commandes de vol électriques, ne permet plus au pilote d'avoir une perception instinctive de l'approche des limites de l'avion (limite en incidence en particulier).

La forme la plus perfectionnée de cette aide consiste à introduire des fonctions de limitation automatique dans le système de commandes de vol, et c'est en définitive bien plus la nature et la qualité de ces fonctions que la qualité du comportement général de l'avion (que l'on sait de toute façon rendre excellente) qui constituent les éléments primordiaux de jugement de bonnes qualités de vol d'un avion à commandes de vol électriques.

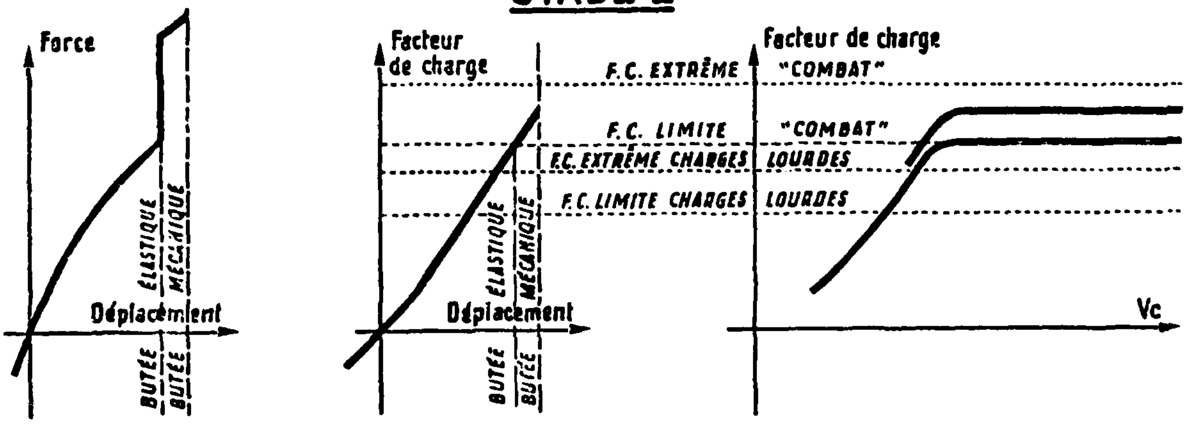
MIRAGE 2000

EVOLUTION DE LA LIMITATION AUTOMATIQUE DU FACTEUR DE CHARGE

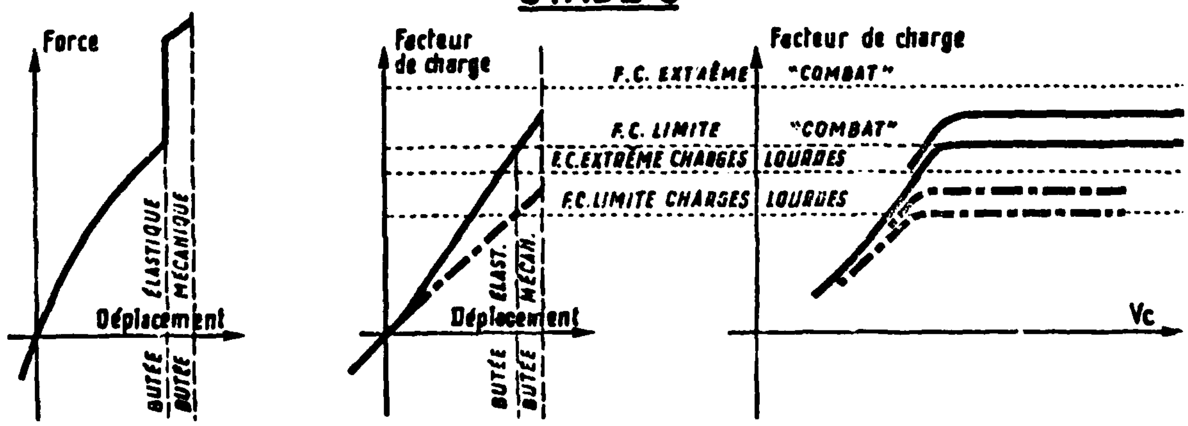
STADE 1



STADE 2



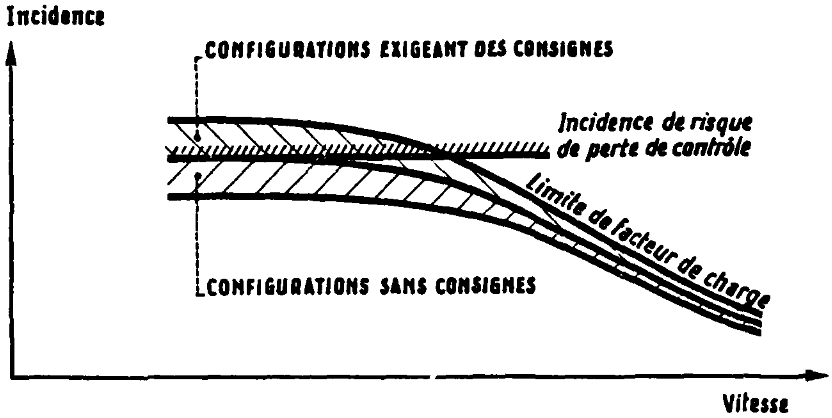
STADE 3



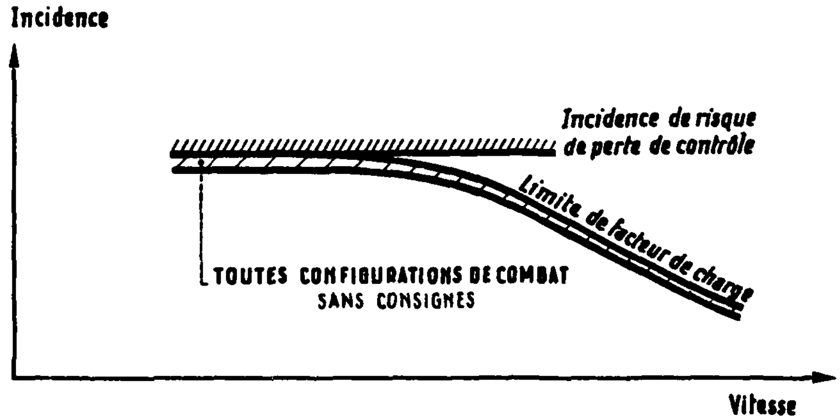
MIRAGE 2000

EVOLUTION DE LA PROTECTION AUTOMATIQUE VIS A VIS DE LA PERTE DE CONTROLE

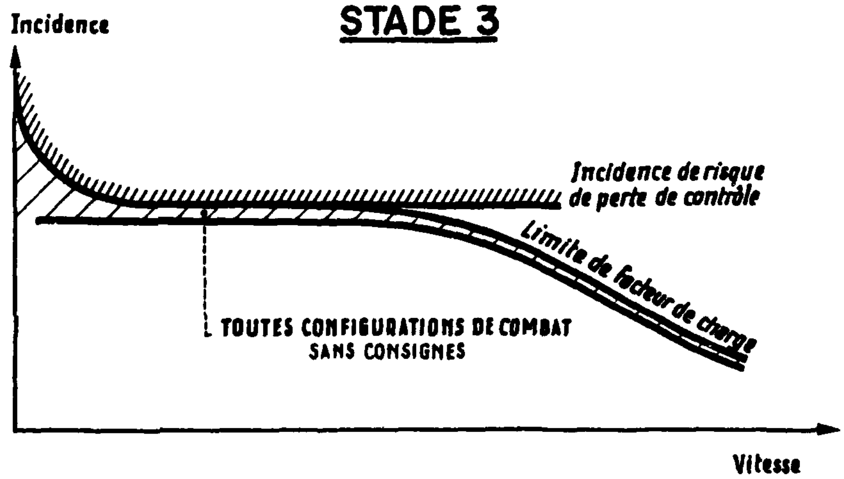
STADE 1



STADE 2



STADE 3



HANDLING QUALITIES OF TRANSPORTS WITH ADVANCED FLIGHT CONTROL SYSTEMS

by

H.A. Mooij and M.F.C. van Gool
National Aerospace Laboratory NLR
Anthony Fokkerweg 2, 1059 CM AMSTERDAM
The Netherlands

SUMMARY

The application of Active Control Technology concepts and in particular the relaxed-static-stability principle applied to future transport aircraft will lead to the introduction of closed-loop primary flight control systems. Adequate handling quality criteria applicable to system design as well as to airworthiness rule-making are required.

Flight simulation and in-flight simulation using mathematical models of aircraft with rate-command/attitude-hold primary flight control systems have been performed for the approach and landing flight phase in order to generate data for criteria development. Configurations featuring in addition blended direct-lift control have been evaluated as well. Based primarily on pilot ratings and commentary, boundaries between "satisfactory" and "acceptable" handling qualities (Cooper-Harper rating 3.5) have been established for a number of criterion formats. While several criteria related to aircraft response are discussed for pitch and roll control, pilot-in-the-loop criteria are proposed for high-precision pitch attitude and altitude control tasks especially.

1. INTRODUCTION

Systematic correlation of pilots' opinions on the aircraft's flying qualities with measured airplane stability and control characteristics was started around 1930(Ref. 1). Somewhat more than ten years later, the first specification for flying qualities based on a series of flight tests of current airplane designs was published by NACA (Ref. 2). This document formed the basis for the first (US) Military Specification with generalized criteria issued in 1943 (Ref. 3). This specification has evolved into the present status, the MIL-F-8785 C (Ref. 4). This specification is basically applicable to aircraft with mechanical primary flight control systems without or with "augmentation" (e.g. rate damping).

With the introduction of closed-loop flight control systems with non-mechanical signal transmission in aircraft, the flying qualities will be determined to a high degree by the characteristics of this system. Unfortunately, the establishment of handling quality criteria for aircraft equipped with such systems is significantly lagging behind the development of flight control system technology. Such criteria are needed as guidelines for adequate system design.

Most flight experience with closed-loop flight control systems in transport aircraft accumulated until now is derived from a limited number of ad-hoc type experiments performed with modified production aircraft and in-flight simulators. The number of reported research programs aimed at the development of generalized handling quality criteria of this category aircraft is disappointingly low. The theme of the present symposium states correctly that the most suitable format of criteria for handling qualities of piloted aircraft depends on the type of control system used in the aircraft. It may be assumed that in many cases the properties of these systems in future transport aircraft will lead in principle to the elimination of the attitude and airspeed stabilization function as part of the task of the pilot. In that situation the remaining control functions to be performed by the pilot will be the establishment of equilibrium states and maneuvering.

A flight control system based on rate-command/attitude-hold for pitch, roll and yaw in combination with an auto-throttle system for airspeed-hold is a logical follow-up of the combination of the Control Wheel Steering (CWS) autopilot mode and an auto-throttle of the more advanced contemporary transports. If part of the existing handling qualities criteria are to remain useful for aircraft fitted with the above indicated systems it is quite reasonable to expect that the elimination of attitude and airspeed stabilization from the piloting task will influence at least the limits/boundaries of the parameters used in existing maneuvering criteria. That in addition new criteria will be needed is easily understood. It is hypothesized that intermittant control, with relatively long periods without any pilot commands leads in general to a more relaxed type of controlling by the pilot. This will lead to a rather critical attitude of the pilot towards the response of motion variables to manipulator deflections. The response is more clearly observable in the aircraft types under consideration here in comparison to contemporary aircraft which have a more pronounced response to atmospheric turbulence.

Moreover, the pilot is aware of the behaviour of the automatic system at the termination of an input. For pitch control this means that "nod-back" at the termination of pitch commands should be within bounds. For roll control the behaviour of the wings-leveller, a system function indispensable for a roll-rate-command/bank-angle-hold system, must be satisfactory.

It would be very useful if the pilot/aircraft closed-loop control structure could be analysed using mathematical models for both the aircraft and the pilot. Although models for the aircraft dynamics are well-known and well-established, models describing the control behaviour of the pilot are still a subject of continuing research. For a mathematical model describing the control behaviour of the human pilot to be useful in handling qualities research, a quantitative and absolute "pilot opinion metric"

This research has been sponsored by the Netherlands Agency for Aerospace Programs (NIVR) and the Department of Civil Aviation (RLD) of The Netherlands.

associated with the model is a requirement. Such a model is at present not available for the airframe/flight control system combination and non-stationary flight phase (flare and landing) under consideration here. Therefore the experimental approach was selected to study the pilot/aircraft combination.

It is considered of importance for the healthy future development of transport aircraft to give serious thought to the lessons learned in the recent past concerning the introduction of modern "fly-by-wire" technology into today's fighter aircraft. It is observed that despite the application of sophisticated design methodology these new aircraft continue to suffer from basic flying qualities deficiencies (Refs. 5, 6). Two of the main points of these lessons as formulated in reference 5 are considered of special importance. One is that in order to achieve the potential of "full-authority augmentation systems", a "logical flying qualities development process must be created which includes sound technical communication among managers, engineers and test pilots". The other one is that, "complexity should be sacrificed for simplicity every time".

In conclusion of this introductory chapter a survey is given of the main programs performed in The Netherlands by the National Aerospace Laboratory (NLR) during the last 10 years.

The NLR activities in the general area of closed-loop flight control systems started in 1972 with a flight test program using an aircraft equipped with a pitch-rate-command/attitude-hold flight control system in combination with a side-stick controller during two-segment noise abatement approaches, reference 7. This was followed in 1974 by a flight simulation program directed at the determination of generalized longitudinal maneuvering criteria for the landing approach and touchdown flight phase (Refs. 8 and 9).

Flight tests on stick force stability as a cue for manual airspeed regulation during approach in aircraft with primary flight control systems featuring pitch attitude-hold, was performed in 1975 and 1976 (Refs. 10, 11, 12) using the NLR-owned Beechcraft Model 80 and the Fokker F-28 prototype.

Flight simulator programs directed at the development of longitudinal (Ref. 13) and lateral-directional (Ref. 14) maneuvering criteria have been performed in 1978 and 1979 using the flight simulator of the NLR.

Selected configurations from the above-mentioned programs were evaluated again in flight using the USAF/Calspan Total In-Flight Simulator (TIFS) during the end of 1980 and the beginning of 1981, reference 15.

The emphasis of the underlying paper is on handling quality criteria related to changing the aircraft attitude and the direction of the velocity vector during the approach and landing based on experiments in the 1978 - 1981 time period.

2. CLOSED-LOOP FLIGHT CONTROL SYSTEMS

2.1 Documented experience

On the basis of the predicted trend in fuel prices it is to be expected that future transport designs will incorporate full-time stability augmentation in order to exploit the gains obtainable from the so-called relaxed-static-stability concept (Ref. 16).

Another development towards the improvement of overall aircraft performance will be the acceptance of the "all-electric aircraft" concept which means that all engine loads, normally associated with the bleed, pneumatic and hydraulic systems are transferred over to the electric power generation system, (Refs. 16, 17 and 18).
These developments will form the powerful drivers towards the introduction of closed-loop flight control systems with non-mechanical signal transmission in transport aircraft. In this section documented flight and flight simulation experience with closed-loop flight control systems will be reviewed.

Observations will first be made with respect to two categories of longitudinal flight control systems:
I: Closed-loop flight control systems which use the horizontal tail as the only aerodynamic control surface.
II: Systems as mentioned under I, which in addition enhance the maneuverability by using lift-modulating aerodynamic surfaces on the wing.
The observations are restricted to programs before 1978, reported in the open literature, in which actual landings were carried out.

Sub I: Closed-loop flight control systems (Refs. 9 and 19 through 26)
Systems based on pitch rate as the primary controlled aircraft motion variable have been favourably commented upon; this holds for flight test as well as flight simulator results. An attitude-hold feature, when available was appreciated.
A system based on normal acceleration as the primary controlled motion variable evaluated in approach and landing (Ref. 23) was not well received by the participating pilots.

Sub II: Closed-loop flight control systems with maneuver enhancement (Refs. 9, 19 and 27 through 29)
In the tests considered, maneuver enhancement resulted in:
- smaller pitch attitude variations during approach path following,
- more precise ILS-glide path tracking,
- suppression of "initial acceleration reversal" at the center of gravity,
- better touchdown position accuracy during flight tests; based on subjective observation (Refs. 28, 29),
- improved control of sink rate at touchdown during flight simulation tests based on objective measurement (Ref. 9).

There are indications that maneuver enhancement can be used to either improve the touchdown position accuracy or to improve the control of sink rate at touchdown.

With respect to lateral flight control systems the following observation is in order. Once an aircraft has a pitch-rate-command/attitude-hold flight control system a roll control system with similar characteristics should be installed for reasons of control harmony and aircraft stabilization. Therefore a roll-rate-command/bank-angle-hold system is the logical choice.

Concerning the documented experience with respect to an appropriate type of pilot's controller for closed-loop flight control systems the following is remarked.
Transport aircraft with closed-loop flight control systems and non-mechanical signal transmission will most probably be equipped with a moving type side stick controller. In a number of the above-mentioned experimental programs such a manipulator has been used (Refs. 9, 21, 23). Without exception favourable opinions were reported in these programs concerning this type of controller in the context of the evaluations performed.

2.2 Selection of flight control systems for the experiments

Based on the observations discussed in section 2.1 the form of rate-command/attitude-hold has been selected for experiments by NLR, the results of which will be discussed in this paper. In addition it is mentioned here that this principle is well-suited for aircraft in which artificial longitudinal stability has taken the place of inherent airframe stability (relaxed-static-stability).
Low-speed handling flight simulator research performed recently as part of NASA's Supersonic Cruise Research program (statically unstable conceptual aircraft) indicates the use of similar concepts, see references 30 and 31.

It was decided to use manual airspeed control in the piloted evaluations. This decision was based on:

- The experience with respect to the combination of a pitch-rate-command/attitude-hold flight control system with manual throttle on the approach during flight test programs (Ref. 12)
- The desire to minimize the number of automatic (sub-)systems required during normal operation as a design philosophy.

For the ground-based simulations (Refs. 13, 14) the mathematical model of an existing medium-weight twin-engined jet transport aircraft was adopted. The combination of horizontal tail area and the center of gravity location were chosen in such a way that a static margin of zero resulted (the horizontal tail area was reduced by 40 %). This mathematical model is called Baseline aircraft A. Starting from this model an aircraft with a 40 % higher gross weight has been defined, called Baseline aircraft B. The latter model was used in relation to criteria development for maneuver enhancement through the application of (blended) direct-lift control.

For both models a primary flight control system was developed in such a way that the dynamics of the aircraft plus flight control system were selectable by changing a limited number of gains and time constants in the mathematical model of the flight control system.

A functional block diagram of the longitudinal flight control system is presented in figure 1. The lateral-directional flight control system is presented in figure 2. In references 13 and 14 the rationale for the flight control system lay-out and the choice of flight control system parameters are discussed in detail.

Next the main characteristics of the flight control systems are described; the way in which the dynamics of the responses to manipulator deflections have been varied is indicated as well.

LONGITUDINAL SYSTEM

The closed-loop formed by the aircraft and the feedbacks of pitch rate and pitch angle functions as a pitch angle stabilizer ("attitude-hold"). Commands generated through manipulator deflections and passed through a low-pass filter and through an element (prefilter) which consists of a proportional and an integral part drive the closed-loop ("rate-command"). For Baseline aircraft B the commands were also passed through a low-pass filter and a wash-out filter in order to deflect aerodynamic surfaces on the wing for direct-lift generation.

The <u>longitudinal short-period frequency</u> was varied in the program through variation of the gain of the pitch attitude and pitch rate feedbacks.

The <u>pitch rate overshoot after a step command input</u> was varied in the program through variation of the relative gains of the proportional and the integral parts of the prefilter (prefilter time constant).

The <u>level of maneuver enhancement</u> was varied in the program through variation of the gain in the path between the manipulator and the direct-lift aerodynamic surfaces.

In addition the longitudinal flight control system features:
- Two-gradient force-versus pitch-rate-command relationship,
- Bank compensation.

LATERAL-DIRECTIONAL SYSTEM

The closed-loop formed by the aircraft and feedbacks of roll rate and bank angle functions as a bank angle stabilizer ("attitude-hold"). Commands generated through manipulator deflections and passed through a lead/lag filter drive the closed-loop formed by the aircraft with roll rate feedback while the bank angle feedback is disabled through logic ("rate-command").

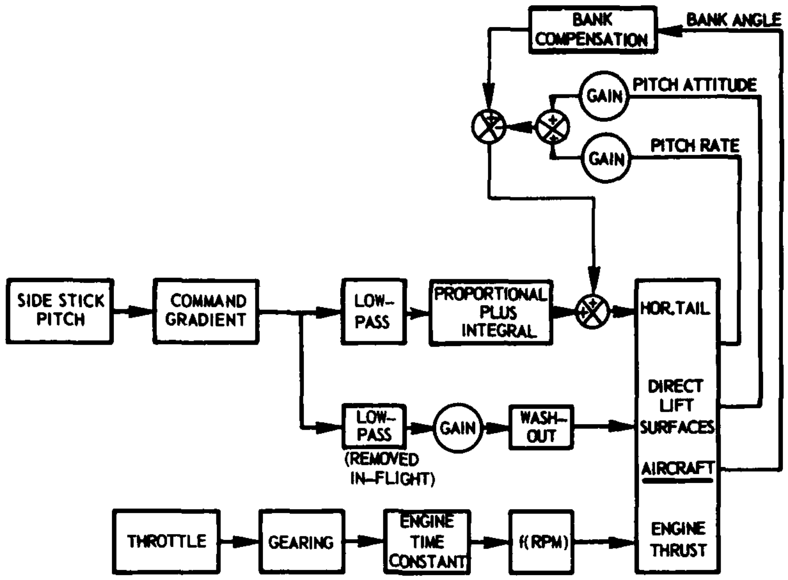


Fig. 1 Block diagram of the longitudinal flight control system.

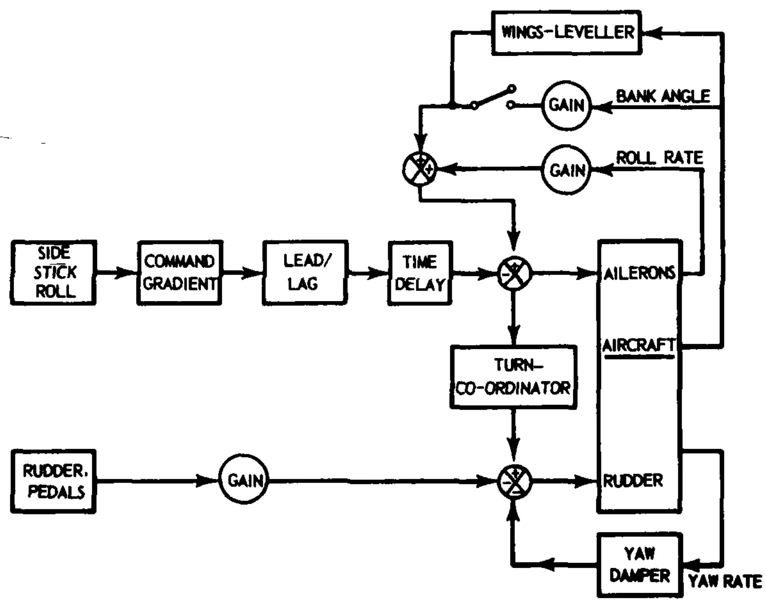


Fig. 2 Block diagram of the lateral-directional flight control system.

The roll mode time constant was varied in the program through variation of the lag time constant of the lead/lag filter.

A pure time delay in the roll command path was varied in the program by delaying the command signal a number of computational steps.

In addition the lateral directional flight control system featured:

- Three-gradient force versus roll-rate command relationship,
- Wings-leveller, active for bank angles between plus or minus 3 degrees,
- Yaw-rate feedback to increase Dutch Roll damping,
- Turn-co-ordinator.

3. EXPERIMENT MECHANIZATION

3.1 Ground-based simulation facility

The ground-based flight simulator used for the experiments is developed and operated by the National Aerospace Laboratory NLR (Fig. 3). At the time of the experiments the equipment included a computer installation, a single seat cockpit with pilot's controllers, flight instruments, a visual system and a motion system.

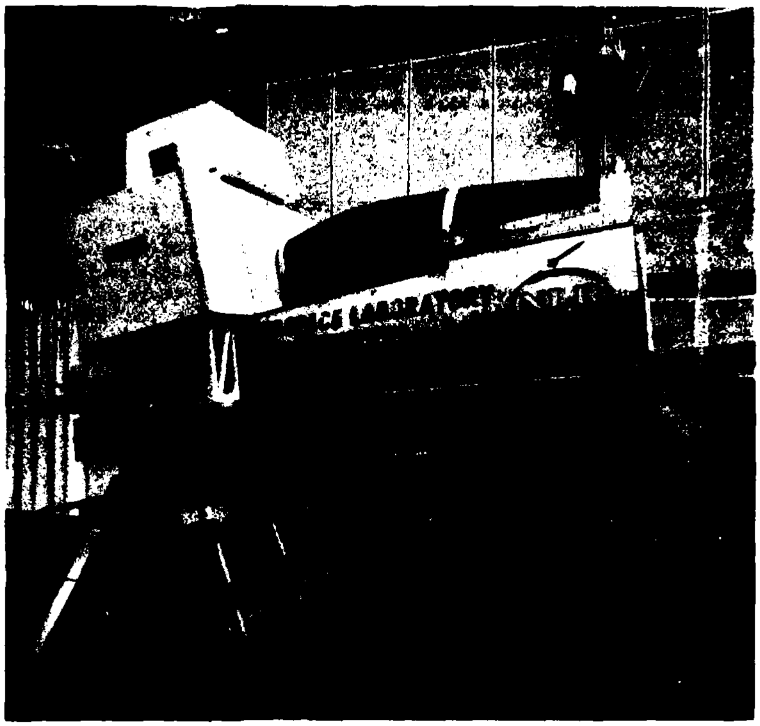


Fig. 3 Cockpit, motion base and visual display system of the NLR flight simulator.

The instrument panel is based on the conventional lay-out for civil aircraft. The pilot's primary instruments are elements of an Integrated Instrument System with an Attitude Director Indicator above a Horizontal Situation Indicator. The pilot's primary (hand) controller was an in-house developed deflection-type side-stick controller. The rudder pedal forces are generated through a control loading system. The (left-hand) throttle was of the type used in fighters.

The visual system consists of a terrain model viewed through a closed-circuit color television system. The visual scene presented to the pilot is collimated to provide images at infinity. The field of view is 43 degrees in the horizontal plane and 16 degrees in the vertical plane.

The motion system has four degrees of freedom corresponding to the following aircraft motions: heave, pitch, roll and yaw. Due to the hydrostatic bearings in the jacks the acceleration noise level and the threshold value of the accelerations are kept very low. High-pass filters for simulation of rotational accelerations and low-pass filters for simulation of specific forces (through tilt-angles) are used.

3.2 In-flight simulation facility

The USAF/Calspan Total In-Flight Simulator (TIFS) was used as the test vehicle in the in-flight validation experiment. A description of the capabilities of this in-flight simulator is given in reference 32. TIFS is a highly modified C-131H (Convair 580) configured as a six degree-of-freedom simulator (Fig. 4).

Fig. 4 USAF-AFWAL/Calspan Total In-Flight Simulator (TIFS)

Computer driven direct-lift flaps, side force surfaces and throttles have been added to yield direct force capability. Moment control is through the elevator, ailerons and rudder. It has a separate evaluation cockpit forward and below the normal cockpit. When flown from the evaluation cockpit in the simulation mode, the pilot control commands are fed as inputs to the model hybrid computer which calculates the aircraft response to be reproduced. These responses, along with TIFS motion sensor signals, are used to generate feedforward and response error signals which drive the six controllers on the TIFS. The result is a high-fidelity reproduction of the motion and visual cues at the pilot's position of the model aircraft.

For this experiment, the evaluation cockpit was set up similar to that in the NLR ground-based simulator. This included the same side-stick controller which was used in the ground simulations. The cockpit side windows were masked to duplicate the peripheral visibility from a typical transport aircraft. The forward field of view was larger than normally encountered in transport aircraft, especially in downward direction. For attitude reference a mark was put on the front screen.

3.3 Dynamics of the configurations

In the ground-based investigation the characteristics of the pitch and roll control systems have been varied in order to generate experimental configurations with varying flying qualities. One configuration having good pitch and roll flying qualities was selected as the base configuration and changes in the characteristics with respect to this base configuration have been investigated. A limited number of descriptors is used to characterize the configurations as indicated in table 1. For the longitudinal dynamics the values of the "equivalent system" parameters are used. The "equivalent system" is the best frequency domain fit of a system with transfer function

$$\frac{q}{s_e} = \frac{K_q\,(s + 1/\tau_q)\,e^{-T_q s}}{(s^2 + 2\zeta_q\,\omega_{n_q}\,s + \omega_{n_q}^{\,2})}$$

to the actual high-order transfer function. Also the values of the normal acceleration sensitivity parameter n_α (normal load factor change per unit change of angle of attack) and of the level of the maneuver enhancement with respect to the maximum level ($K_{DLC}/K_{DLC_{max}}$) are presented in table 1. The time constant of the wash-out filter has been kept constant at 5 s.
For the roll dynamics the equivalent roll mode time constant τ_R' and the equivalent time delay T_p are presented. These parameters are defined using the roll rate response to a step-type roll command input as indicated in figure 5.

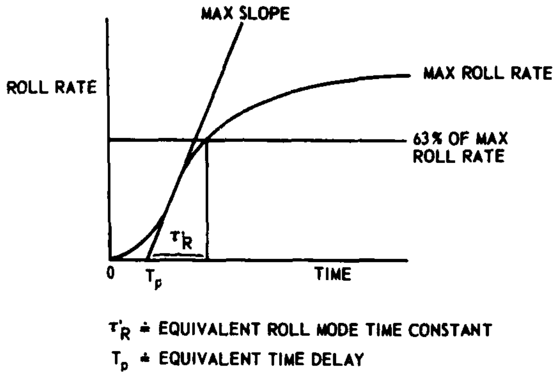


Fig. 5 Definition of equivalent roll mode time constant and equivalent time delay, using the roll rate response to a step-type control input.

As can be observed in table 1 five types of variations have been realized.
A selection was made from these configurations to be validated in the in-flight investigation. The base-line configuration and one configuration out of every group was selected. The 7 configurations selected for in-flight validation will be characterized by the acronyms used in the ground-based investigation:

E-5/T-1: base configuration (pitch and roll)
E-1 : low longitudinal short-period frequency
F-1 : large pitch rate overshoot
G-1 : low n_α; no DLC
G-3 : low n_α + DLC
T-4 : large roll mode time constant
T-7 : roll command time delay

The dynamics of the computer models used ground-based and in-flight were identical, as was verified by comparing responses to step-type control inputs. Assuming no difference in instrument dynamics ground-based and in-flight, the configuration dynamics as observed during flight on instruments were identical.
The visual system used in the ground-based investigation and the model-following system in the in-flight investigation introduced additional dynamics such that in some respects the information as obtained from the outside visual scene differed between ground-based and in-flight investigations. This difference will be taken into account when analyzing the results.

3.4 Experiment design

The piloting task consisted of performing approaches in IMC conditions, using only raw ILS information for guidance followed by a VMC segment below 91 m (300 ft) terminated by a landing. Offsets in the glide path and localizer indications were removed at certain points during the approach, thus forcing the pilots to maneuver the aircraft, thereby facilitating their evaluation of the configuration's handling qualities.

In both ground-based and in-flight investigations one configuration was evaluated per session in which three to five approaches and landings were carried out. Ample time was devoted to familiarization. In both types of investigations three pilots have evaluated all configurations.

In the ground-based investigation the environmental conditions consisted of wind shear and light to moderate turbulence with mean wind at runway level in runway direction for the longitudinal investigations and 15 kts crosswind in the lateral-directional investigations. In the in-flight investigation the existing natural crosswinds were cancelled for the longitudinal configurations and for the lateral-directional evaluation flights cross-winds of 10 to 15 kts were simulated. Natural turbulence was used to disturb the model and where necessary artificially generated turbulence was introduced into the model.

Each pilot received a written briefing guide and rating information. Before flying they were verbally briefed on basic experiment purposes and simulation procedures. In all experiments the pilots have evaluated all configuration in a randomized order. To minimize carry-over effects (usage of experience obtained in one configuration during the next), averaged results of the three pilots have been analyzed.

4. CRITERIA FOR LONGITUDINAL MANEUVERING

4.1 Results of experiments

Three groups of configurations (E, F and G) are discussed. The Cooper-Harper ratings given for the configurations with varying short-period frequency are presented in figure 6. In this figure the results for the in-flight configuration E-1 is compared to the result for the ground-based configuration E-2 for reasons explained later.

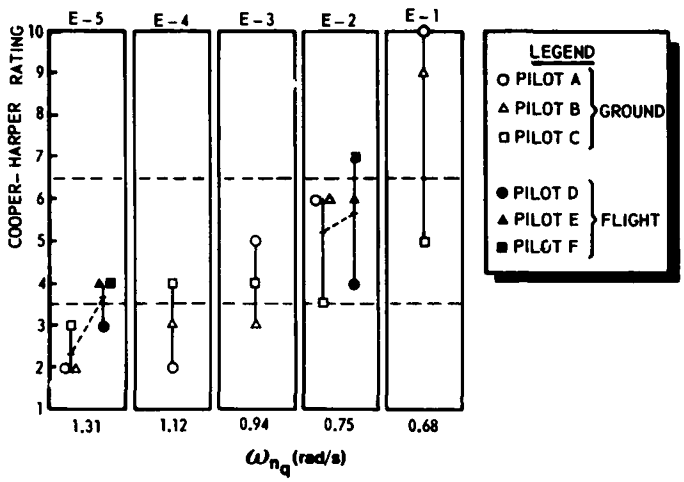


Fig. 6 Cooper-Harper ratings for the configurations with different short-period frequency.

The degradation of handling qualities with decreasing value of the short-period frequency is clearly indicated both ground-based and in-flight. Configuration E-5 was rated 3-4 on the Cooper-Harper scale in the flight evaluations while ratings of 2-3 were obtained during the evaluations on the ground. According to pilot commentary obtained in flight, a relative high level of attention was required for air-speed control in the in-flight simulator. Two pilots stated that for this reason they rated the configuration in the unsatisfactory region. They stated that when the speed regulation task would be eliminated they would certainly end up in the satisfactory region (Cooper-Harper ratings of 2 and 3 were mentioned).

The Cooper-Harper ratings given for the configurations with varying pitch rate overshoot after a step command input are presented in figure 7.

The result for the in-flight configuration F-1 is not directly comparable to the result for the ground-based configuration F-1 for reasons explained later. In figure 7 an additional "F-1/flight" has been introduced for a value of τ_q somewhat higher than the value existing in the ground-based experiment. The result obtained in flight is in concurrence with the trend indicated by the ground-based results.

The Cooper-Harper ratings given for the configurations with varying degree of maneuver enhancement are presented in figure 8. The rating for configuration G-1 for pilot D is considered an outlier and is not considered further (see reference 15). For configuration G-3 the mean of the ratings in flight was higher than the mean of the ratings on the ground. Two of the pilots in the in-flight evaluation complained explicitely about disturbing heave motion associated with stick motions. This was their prime reason not to rate the configuration better. Non-intentional amplification of the high-frequency content for the normal acceleration at the pilot seat of the model output by the model following system has been established. No proper judgement of this configuration is possible therefore. It is of interest to mention that the in-flight measured vertical speed at touchdown reduced substantially through addition of maneuver enhancement from a mean value of 0.7 m/s for configuration G-1 to 0.3 m/s for configuration G-3.

Remarks concerning the dynamics of the visual system of the flight simulator in relation to the dynamics of the model-following system of the in-flight simulator are in order here.

In the above presented comparison for the E- and F-configurations the slight differences due to the different additional dynamics of the visual system and model-following system have been taken into account, thus emphasizing the comparison for the visual flight phase.

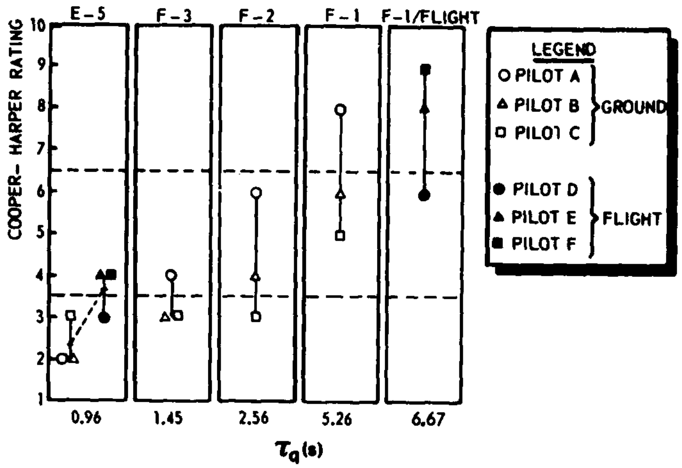


Fig. 7 Cooper-Harper ratings for the configurations with different levels of pitch rate overshoot.

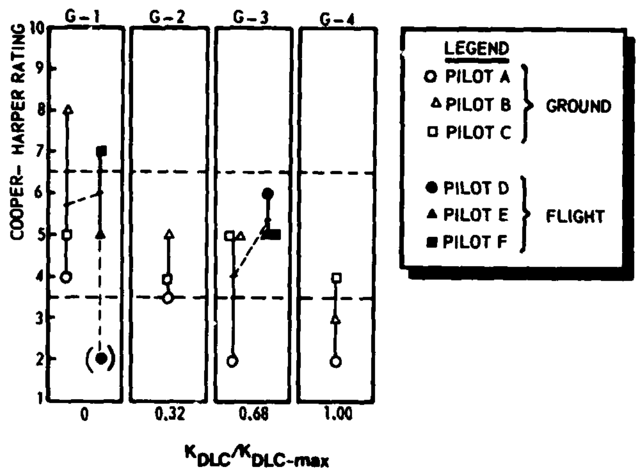


Fig. 8 Cooper-Harper ratings for configurations with different levels of maneuver enhancement.

When the pilot was using the outside visual scene, the dynamics of the visual system of the flight simulator and the dynamics of the model-following system of the in-flight simulator are superimposed on the model dynamics. The dynamics of the visual system of the flight simulator can be characterized as a time delay of 0.05 s and a first order low-pass filter with a time constant of 0.08 s. The effect of these additional elements can be expressed as an increase of the value of the equivalent time delay with 0.13 s, resulting in T_q = 0.2 s.

Analysis of the pitch rate response to a step-type command input of the in-flight simulator suggests that the model-following system can be represented by a time delay of 0.07 s and some additional "lead" to quicken the response after the delay.

With respect to the comparison ground versus flight the additional "lead" of the model following system warrants the conclusion that:

Configuration E-1/flight should be compared to configuration E-2/ground
Configuration F-1/flight should be compared to a configuration with a pitch rate overshoot larger than existing for configuration F-1/ground. An additional configuration "F-1/flight" with τ_q = 6.67 is therefore incorporated in figure 7.

Reference 15 discusses these matters in detail.

In conclusion of this section an observation related to the heave response of the motion system during ground-based simulation and of the model-following system during in-flight simulation is necessary for the configuration with maneuver enhancement (configuration G-3).

In the ground-based simulator the heave response is attenuated by the filters used in the command path to the motion system; in principle this can mask a problem area which can develop in real flight. After close examination of in-flight recorded data it was established that the calibration step responses for the in-flight simulator indicated a non-intentional initial overshoot of the normal acceleration response at the pilot station. Amplification of the high frequency content of the model output by the model following system is apparent on the flight records of approaches flown. A flexible mode of the TIFS airframe plays a role in the observed phenomenon. These observations have led the authors to the conclusion that it cannot be determined how much the pilot complaints concerning the "disturbing" heave motions mentioned before are related to the nominal configuration dynamics.

4.2 Criteria

Both open-loop and pilot-in-the-loop maneuvering criteria have been evaluated on the basis of experimental results.

OPEN-LOOP CRITERIA

On the basis of the Cooper-Harper ratings of the flight simulator investigation, the correlation with the following criteria has been analyzed:
- -C*-criteria, reference 33
- -Large advanced supersonic aircraft criterion, reference 34
- -US Military Specification short-period response criterion, reference 4
- -Criterion on the compatibility of steady maneuvring forces and pitch sensitivity, reference 35

The flight simulator results have shown that the first two criteria are not applicable to the type of control system under consideration (Ref. 13).

Analysis of the MIL-F-8785 C short-period response criterion, when interpreted through the use of equivalent system parameters (ω_{n_q} versus n_{α_e} ; $n_{\alpha_e} = \frac{V}{g} \cdot 1/\tau_q$.) led to the conclusion that this criterion has shortcomings when used for aircraft with the flight control systems considered here. The criterion allows lower ω_{n_q} values than the results indicate, while also a configuration characterized here as having too much overshoot after a step command input, configuration F-2 (CH=4.3) lies well inside the level 1 area, as is shown in figure 9.

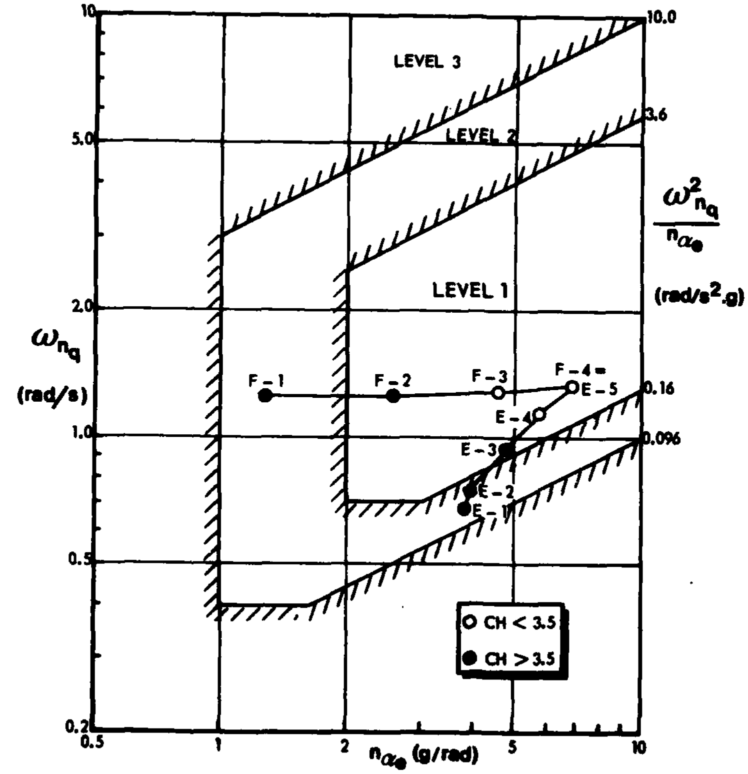


Fig. 9 Flight simulation results versus US Mil Spec. short-period response criterion.

One of the drawbacks of the criterion is considered to be the fact that a criterion for damping ratio is specified separately. Especially in the area of relative low short-period undamped natural frequencies (vicinity of the lower boundary for $\omega^2_{n_q}/n_{\alpha_e}$) the effects of the numerator time constant, undamped natural frequency and damping ratio should be considered simultaneously.

A proposed criterion which takes the effect of these three parameters into account while it incorporates the effect of time delays as well is the "rise time" of the pitch rate response to a step command input.

Rise time (T_{rise}) as defined here: the time in which the pitch rate response to a step-type control input reaches 90 percent of the (final) steady state value. The proposed criterion is:

	Level 1	Level 2
T_{rise}	< 1.0 s	< 1.7 s

[Although time delay was not varied in the experiments, it is possibly of interest to observe that an equivalent time delay of 0.2 s in combination with the equivalent system parameters ω_{n_q}, ζ_q and τ_q of configuration E-5 has not degraded pilot opinion in the ground-based experiments].

Settling time of the pitch rate response to a step command input is also considered of importance and possibly usable as criterion. Settling time (T_{settle}) as defined here: the time after which the pitch rate response remains within a band of values which range from 90 percent to 110 percent of its (final) steady state value. In the determination of T_{settle} the effect of low-frequency dynamics should be eliminated. A second-order equivalent system description is one means to generate the required time responses. Pilot evaluation of configurations with various values of the equivalent damping ratio, ζ_q as well as the equivalent undamped natural frequency ω_{n_q} is required to propose a criterion for T_{settle}.

The format of the fourth criterion mentioned in the beginning of this section, "criterion on the compatibility of steady maneuvering stick forces and pitch sensitivity", is considered especially appropriate in the area of high levels of pitch rate overshoot after a step command (e.g. large lead in the flight control system, F-configurations), see also reference 36.

A criterion format proposed in a document discussing proposed revisions to MIL-F-8785 B (ASG) (Ref. 35) is directed at the compatibility of appropriate control gains for "stick force per g" and pitch acceleration. These two control gains can become incompatible in the sense that the pitch control sensitivity is too high relative to the "stick force per g". The criterion puts limits to the expression $\left|\frac{F_e}{n_p}\right|_{ss} \times \left|\frac{\ddot{\theta}}{F_e}\right|_{max}$.

This parameter can be considered as an appropriate description for the Control Anticipation Parameter (CAP, reference 37) for aircraft for which the pitch rate to stick force transfer function deviates appreciably from classical (bare airframe) expressions due to their flight control systems. For the type of control system considered, side-stick controlled pitch-rate-command (plus attitude-hold), the following criteria have been established for the approach and landing flight phase:

	Level 1	Level 2
$\left\|\frac{F_e}{n_p}\right\|_{ss} \times \left\|\frac{\ddot{\theta}}{F_e}\right\|_{max}$	$< 0.7\ \text{rad/s}^2.\text{g}$	$< 2.9\ \text{rad/s}^2.\text{g}$

These limiting values for $\left|\frac{F_e}{n_p}\right|_{ss} \times \left|\frac{\ddot{\theta}}{F_e}\right|_{max}$ (proposed in the present paper) deviate, however, considerably from the values published in reference 35. When attention is focussed on the substantiation of the criterion for Category C flight phase in that reference it has to be deduced that no data at all have been available to substantiate the limit values published. These values are the same as the numerical values of the upper limit of the CAP of the present Military Specification (Ref. 4). Also the upper limits for Level 1 and 2, "terminal flight phase" for $\left|\frac{F_e}{n_p}\right|_{ss} \times \left|\frac{\ddot{\theta}}{F_e}\right|_{max}$ published in recommended flying qualities criteria for supersonic transports (Ref. 38), *although more restrictive than the values for CAP of the* Military Specification, are not in correspondence with the experimental results obtained here.

Alternative approaches to define the CAP based on time history measurements, are discussed in references 38, 39, 40 and 41.

PILOT-IN-THE-LOOP CRITERIA

On the basis of the Cooper-Harper ratings of the flight simulator investigation the correlation with the following criteria has been analyzed:
- Simplified pitch dynamic response criterion, reference 35
- Inferred closed-loop criterion, reference 42
- Original pitch dynamic response criterion, reference 43.

Although closely related to the "original pitch dynamic response criterion", the "simplified pitch dynamic response criterion" has shown to be less useful in practice as is discussed in reference 13.

No correspondence between the "inferred closed-loop criterion" and the experimental results have been shown, reference 13. The "original pitch dynamic response criterion" or Neal-Smith criterion is clearly the most versatile. Applying the value for "minimum bandwidth" as is proposed in reference 35 for transport aircraft in terminal flight phases, $\omega_{BW-\theta} = 1.2$ rad/s, the result of ground-based simulation and in-flight simulation has indicated that for Level 1 handling qualities the following criteria should apply:

$$\left|\frac{\theta}{\theta_c}\right|_{max} < 0\ (\text{dB}) \text{ and } \angle_{pc} < +45\ (\text{deg})$$

The limit value on resonance deviates appreciably from the boundaries of the original pitch dynamic response criterion. In figure 10 the experimental results are plotted. The dynamics of the visual system (of the flight simulator) are incorporated in the transfer function used to calculate resonance and pilot lead.

With respect to aircraft deficient in flight path response and for that reason incorporating maneuver enhancement as part of the flight control system, no criteria are in existence. It is not possible to give an "open-loop" type of criterion to specify the amount of maneuver enhancement through blended DLC required for aircraft deficient in flight path response (n_α too low), reference 13. The reason is that altitude loop performance obtainable depends among others on pilot compensation used in the inner loop (attitude). As a possible format for a criterion, a series closure structure is envisaged in which the innerloop is closed according to the principle discussed above ("minimum bandwidth" 1.2 rad/s, while observing the $|\theta/\theta_c|_{max}$ and $\angle_{pc}$ boundaries) and the outerloop is closed (with only a gain to represent the pilot action) in such a way that a phase margin of 30 degrees is obtained. The proposed criterion, see figure 11, is the "minimum bandwidth", ω_{BW-h}, which should result after the loop closure. Based on flight

simulator evaluations only, the criterion for Level 1 handling qualities proposed is:

$$\omega_{BW-h} > 0.55 \text{ rad/s}$$

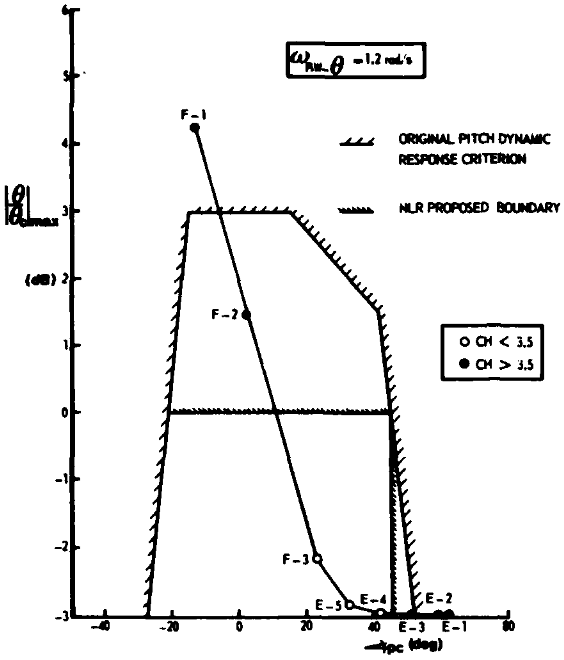


Fig. 10 NLR proposed boundary in the $|\theta/\theta_c|_{max} - \measuredangle_{pc}$ plane; flight simulation results.

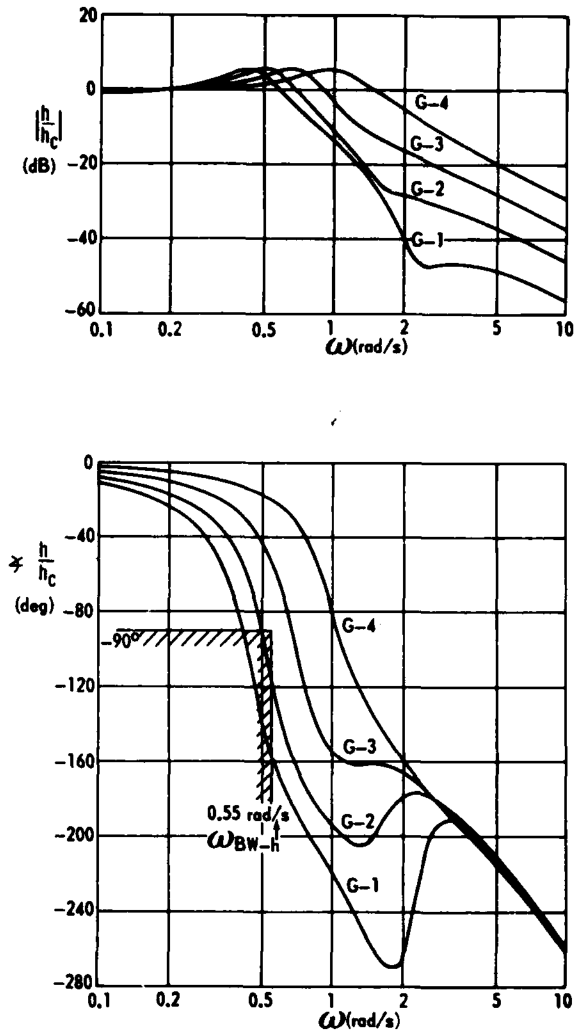


Fig. 11 NLR proposed criterion with respect to maneuver enhancement (Ref. 13).

5. CRITERIA FOR LATERAL MANEUVERING

5.1 Results of experiments

Two groups of configurations will be presented. The Cooper-Harper ratings given for the configurations, for which the equivalent roll mode time constant was the parameter varied, are presented in figure 12. The degradation of handling qualities with increasing values of the equivalent roll mode time constant is clearly observable both ground-based and in-flight. However, a value of 2 seconds being rated unacceptable in the ground-based experiments (average Cooper-Harper rating 7.3) is rated unsatisfactory but acceptable in-flight (average Cooper-Harper rating 5.7). The Cooper-Harper ratings given for the configurations, for which the equivalent time delay in the roll command path was the parameter varied, are presented in figure 13.

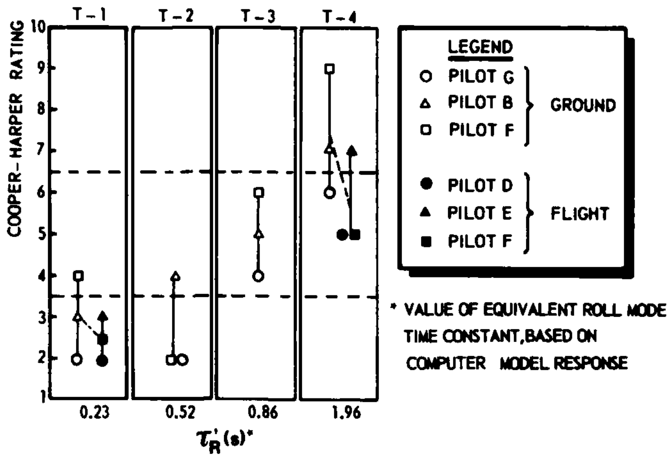


Fig. 12 Cooper-Harper ratings for the configurations with different equivalent roll mode time constant.

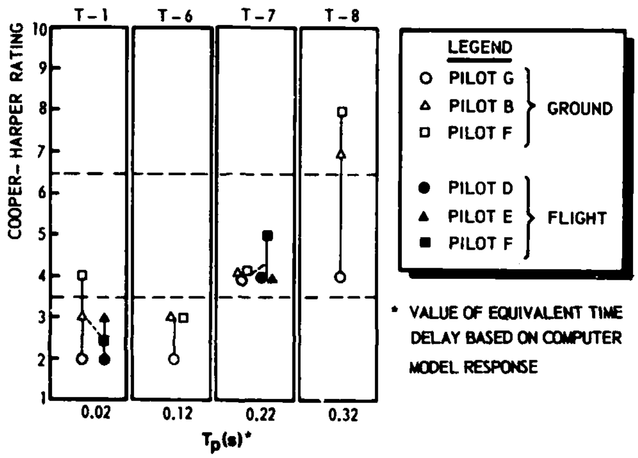


Fig. 13 Cooper-Harper ratings for the configurations with different equivalent time delays in the roll command path.

Degradation of handling qualities with increasing values of equivalent time delay is very similar ground-based and in-flight.

It must be emphasized that this comparison is based on equivalent roll mode time constant and equivalent time delay as determined from the computer model step responses. The pilot commentary indicated, however, that the Cooper-Harper ratings were largely determined by the impressions that were obtained in the final approach and landing (visual segment). In that situation the dynamics of the visual system of the ground-based simulator and of the model-following system of the in-flight simulator are superimposed on the model dynamics. The dynamics are not the same. The ground-based visual system can be approximated by a small time delay of 0.03 s and a first-order low-pass filter with a time constant of 0.08 s, thus increasing both equivalent time delay and equivalent roll mode time constant. The roll response of the in-flight simulator suggests that the model-following system can be represented by a time delay of 0.12 s and "some additional lead", to quicken the response after the delay. The effect is an increase in equivalent time delay and a decrease in equivalent roll mode time constant.

5.2 Criteria

In the ground-based investigation the correlation of the Cooper-Harper ratings with a large number of criteria parameters based on computer responses has been analyzed. Parameters included bank angle in one second, time to reach 30 degrees bank angle, maximum available roll rate, maximum available roll acceleration, equivalent roll mode time constant, equivalent time delay etc. The result was that equivalent roll mode time constant and equivalent time delay of the computer model showed the best correlation. In reference 14 it is shown that a regression equation of the following form

$$CH_{est} = 1.6 + 2.7\ \tau_R' + 10\ T_p$$

predicted the ratings for all twelve configurations evaluated in two flight simulator experiments with

good success (All actual ratings were within one rating unit from the estimate). The regression equation does not include the effect of the visual system dynamics. If the parameters are modified as follows:

$\tau'_{R_v} = \tau'_R + 0.08$ and $T_{p_v} = T_p + 0.03$, the regression equation changes into $CH_{est} = 1.1 + 2.7\,\tau'_{R_v} + 10\,T_{p_v}$.

Using actually measured equivalent roll mode time constant and equivalent time delay, this equation predicts the following ratings for the three configurations that have been evaluated in-flight:

Conf. T-1, $CH_{est} = 2.9$ ($CH_{actual} = 2.5$)

Conf. T-4, $CH_{est} = 8.0$ ($CH_{actual} = 5.7$)

Conf. T-7, $CH_{est} = 5.1$ ($CH_{actual} = 4.3$)

The fact that the actual rating for configuration T-4 was significantly lower than predicted indicates that probably the coefficient of τ'_{R_v} is a little too high.

In MIL-F-8785 C (Ref. 4), no link is put between allowable roll mode time constant and roll time delay.
For Class II and III aircraft (medium to heavy-weight transport) a boundary on roll mode time constant reads:
Level 1 $\tau_R < 1.4$ s
For time delay a general boundary is mentioned
Level 1 $T_p < 0.1$ s
The modified regression equation mentioned can be used to validate these boundaries assuming the other parameter is kept at some specified value. Assuming that time delay is negligible ($T_{p_v} \approx 0$) the criterion on roll mode time constant for Level 1 handling qualities is:

$$\tau'_R < 0.9 \text{ s},$$

which is a value that is smaller than the MIL-F-8785 C criterion. Assuming that the roll mode time constant is small (e.g. $\tau'_{R_v} \approx 0.3$ s) the criterion on time delay for Level 1 handling qualities is:

$$T_{p_v} < 0.16 \text{ s}.$$

This value is considerably larger than the value of 0.1 s mentioned in MIL-F-8785 C (Ref. 4) but comparable to the boundary of 0.17 s mentioned in reference 38.
The equation suggests that for larger equivalent roll mode time constants, the boundary on allowable equivalent time delay becomes more restricted.

These results seem to indicate that limitations on rise time of roll rate to a step-type command input may have merit as a criterion.

6. CONCLUDING REMARKS

Ground-based and in-flight experiments have been carried out to investigate the pilot opinion on handling qualities of transport aircraft equipped with side-stick controlled rate-command/attitude-hold flight control systems. The dynamics of the pitch and roll control system have been varied such that boundaries in handling quality criteria could be established.

The in-flight simulation program has been extremely valuable in supporting most of the results obtained during ground-based simulation programs and in providing new insight. In the in-flight experiments a higher piloting effort for airspeed regulation became apparent, whereas large roll mode time constants degraded the pilot opinion less than in the ground-based experiments. A general conclusion is that in order to obtain "satisfactory" pilot opinions these aircraft must possess a somewhat higher equivalent short-period undamped natural frequency and a smaller equivalent roll mode time constant than those allowable for contemporary transports with conventional flight control systems.

More specifically the observations with respect to the longitudinal handling quality criteria are as follows:
- When comparing the values of equivalent system parameters of the investigated configurations with the MIL-F-8785 C short - period response criterion, it appears that the minimum value of short-period undamped natural frequency is too lenient.
  A more comprehensive criterion applicable in this area which incorporates the effects of the numerator time constant, the undamped natural frequency and the damping ratio simultaneously as well as the equivalent time delay, is the "rise time of the pitch rate response to a step command input". Limit values for Level 1 and Level 2 flying qualities have been proposed.
- Limit values for Level 1 and Level 2 flying qualities have been proposed for a criterion concerning the compatibility of steady maneuvering stick forces and pitch sensitivity which is appropriate for configurations with high levels of pitch rate overshoot after a step command input.
- It is shown that a pilot-in-the-loop criterion based on limiting closed-loop resonance and pilot-lead compensation has merit for pitch control of transport aircraft in the approach and landing flight phase. Based on a "minimum bandwidth" of 1.2 rad/s a tentative boundary for Level 1 flying qualities is presented.
- Based only on the ground-based experiments a criterion has been specified concerning the minimum amount of maneuver enhancement for aircraft deficient in flight path response to attitude changes (n_α too low). The criterion uses the minimum bandwidth of the outer-loop obtainable after sequential closure of attitude and altitude loops.

The observations with respect to the lateral handling qualities are as follows:
- For negligible equivalent time delays the maximum allowable value for the equivalent roll mode time constant is smaller than the value published in MIL-F-8785 C.
- The allowable equivalent time delay in combination with a small value of the equivalent roll mode time constant is larger than the value published in MIL-F-8785 C, which is, however, mainly based on fighter experience.

REFERENCES

1 Perkins, C.D., Development of Airplane Stability and Control Technology, Journal of Aircraft, Vol. 7, No. 4, July - August 1970, pp. 290-301.

2 Gilruth, R.R., Requirements for satisfactory flying qualities of airplanes, 1943, NACA TR No. 755.

3 Army Air Forces Specification No. C-1815, (also Navy BuAer SR-119), "Stability and control requirements for airplanes", 31 August 1943.

4 Military Specification, Flying Qualities of Piloted Airplanes, MIL-F-8785 C, November 1980.

5 Smith, R.E. Evaluating the flying qualities of today's fighter aircraft, "Combat aircraft maneuverability", 1982, AGARD Conference Proceedings No. 319, pp. 4/1 - 4/9.

6 Berry, D.T., Flying qualities criteria and flight control design, 1981, AIAA Paper 81-1823.

7 Mooij, H.A., Flight experience with an experimental electrical pitch-rate-command/attitude-hold flight control system, "Advances in control systems", 1974, AGARD-CP-137, pp. 24/1-24/8.

8 Mooij, H.A., Handling quality criteria development for transport aircraft with fly-by-wire primary flight control systems, "Impact of active control technology on airplane design", 1975, AGARD-CP-157, pp. 8/1-8/14.

9 Mooij, H.A., Handling quality criteria development for transport aircraft with fly-by-wire primary flight control systems, 1976, NLR TR 74141 U.

10 Mooij, H.A., and Gool, M.F.C. van, The need of stick force stability for attitude-stabilized aircraft, Part I, 1976, NLR TR 76125 U.

11 Mooij, H.A., and Gool, M.F.C. van, The need of stick force stability for attitude-stabilized aircraft, Part II, 1977, NLR TR 77027 U.

12 Mooij, H.A., and Gool, M.F.C. van, Flight test of stick force stability in attitude-stabilized aircraft, Journal of Aircraft, Vol. 15, No. 9, September, 1978, pp. 562-566.

13 Mooij, H.A., Boer, W.P. de, Gool, M.F.C. van, Determination of low-speed longitudinal manoeuvring criteria for transport aircraft with advanced flight control systems, 1979, NLR TR 79127 U.

14 Gool, M.F.C. van, Roll response criteria for transport aircraft with advanced flight control systems in the landing approach and touchdown, 1980, NLR TR 80103 U.

15 Gool, M.F.C. van, Low-speed handling qualities of advanced transport aircraft: a comparison of ground-based and in-flight simulator experiments, 1982, NLR TR (forthcoming).

16 Swihart, J.M., The next generation of commercial aircraft - the technological imperative, 1980, ICAS, 12th Congress.

17 Cronin, M.J., The all-electric airplane as an energy efficient transport, 1980, SAE Paper 801131.

18 Bird, D.K. The All-Electric Aircraft, 1980, ICAS Proceedings 12th Congress, p.p. 189-201.

19 Flora, C.C., Kriechbaum, K.L., Willich, W., A flight investigation of systems development for reducing pilot workload and improving tracking accuracy during noise-abatement landing approaches, 1969, NASA CR-1427.

20 Staff of NASA Langley R.C., Determination of flight characteristics of supersonic transports during the landing approach with a large jet transport in-flight simulator, 1967, NASA TN D-3971.

21 Lair, D. and Larson, H.B., Military transport (C-141) Fly-by-wire program. Volume I: Control development, system design and piloted simulation evaluation. Volume II: Hardware, installation and flight test, 1974, AFFDL TR-74-52 Vol. I and Vol. II.

22 Grantham, W.D., Nguyen, L.T., Deal, P.L., Neubauer, Jr., M.J. Smith, P.M., Gregory, F.D., Ground based and in-flight simulator studies of low-speed handling characteristics of two supersonic cruise transport concepts, 1978, NASA TP 1240.

23 Cazenave, A., and Irvoas, J., Resultats relatifs à l'experimentation sur simulateur et en vol d'un système de commande de vol electriques généralisables, "Stability and Control", 1979, AGARD-CP-260, p.p. 6/1-6/14.

24 Urie, D.M., L-1011 Active controls design philosophy and experience, "Stability and Control", 1979, AGARD-CP-260, p.p. 20/1-20/9.

25 Johnston, J.F., and Urie, D.M., Development and flight evaluation of active controls in the L-1011, 1978, NASA CP 2036.

26 Urie, D.M., Piloted flight simulation for active control design development, 1978, AIAA Paper 78-1553.

27 Quigley, H.C., Snyder, C.T., Fry, E.B., Power, L.J., Innis, R.C., Latham, W., Flight and simulation investigation of methods for implementing noise-abatement landing approaches, 1970, NASA TN D-5781.

28 Lykken, L.O., and Shah, N., Direct lift control for improved automatic landing safety and performance for a large transport aircraft, 1971, AIAA Paper No. 71-906.

29 Flapper, J.A., Throndsen, E.O., L-1011 flight control system, 1977, AGARD-AG-224.

30 Feather, J.B., and Joshi, D.S., Motion-base simulator results of advanced supersonic transport handling qualities with active controls, 1981, AIAA Paper 81-1819.

31 Averett, B.T., Simulator investigation of arrow-wing low-speed handling qualities, 1979, NASA CP 2108.

32 Reynolds, P.A., Wasserman, R., Fabian, G.J., Motyka, P.R., Capability of the Total In-Flight Simulator (TIFS), 1972, AFFDL-TR-72-39.

33 Tobie, H.N., Elliott, E.M., and Malcom, L.G., A new longitudinal handling qualities criterion, 1966, National Aerospace Electronics Conference (NAECON).

34 Sudderth, R.W., Bohn, J.G., Caniff, M.A., Bennett, G.R., Development of longitudinal handling qualities criteria for large advanced supersonic aircraft, 1975, NASA CR-137635.

35 Chalk, C.R., DiFranco, D.A., Lebacqz, J.V., Neal, T.P., Revisions to MIL-F-8785 B (ASG) proposed by Cornell Aeronautical Laboratory, 1973, AFFDL-TR-72-41.

36 Mitchell, D.G., and Hoh, R.H., Low-order approaches to high-order systems: problems and promises, 1981, AIAA Paper 81-1774.

37 Bihrle, W. Jr., A handling qualities theory for precise flight path control, 1966, AFFDL-TR-65-198.

38 Chalk, C.R., Calspan recommendations for SCR flying qualities design criteria, 1980, Calspan Report No. 6241-F-5 (NASA CR-159236).

39 A'Harrah, R.C., Hodgkinson, J., and LaManna, W.J., Are today's specifications appropriate for tomorrow's airplanes, "Stability and Control", 1979, AGARD-CP-260, p.p. 23/1-23/12.

40 A'Harrah, R.C., and Woodcock, R.J., The military flying qualities specification, a help or a hindrance to good fighter design?,"Combat aircraft maneuverability", 1982, AGARD-CP-139, p.p. 7/1-7/13.

41 Stengel, R.F., A unifying framework for longitudinal flying qualities criteria, 1981, AIAA Paper 81-1889.

42 Ashkenas, I.L., Hoh, R.H., Graig, S.J., Recommended revision to selected positions of MIL-F-8785 B (ASG) and background data, 1973, AFFDL-TR-73-76.

43 Neal, T.P., and Smith, R.E., An in-flight investigation to develop control system design criteria for fighter airplanes, Vol. 1, 1970, AFFDL-TR-70-74.

TABLE 1

Aircraft/flight control system parameters derived from the computer model for the configurations investigated

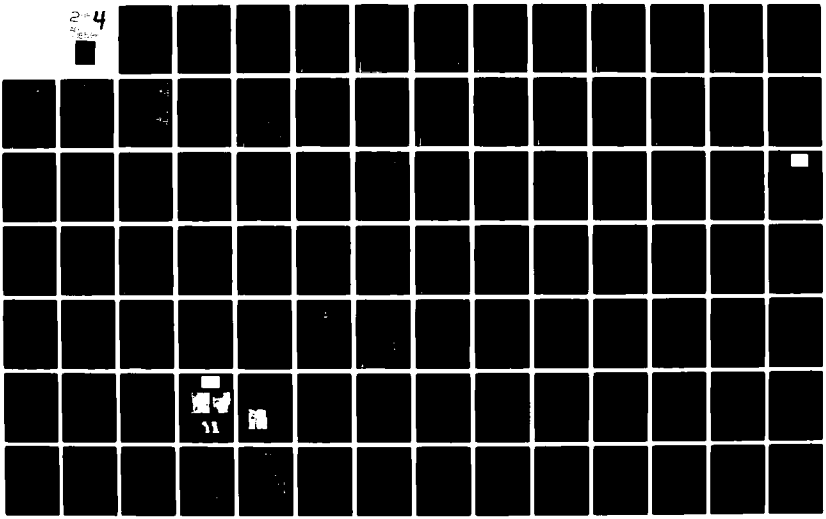
Parameter varied	Config.	Equivalent system parameters (pitch) ω_{n_q} (rad/s)	ζ_q	τ_q (s)	T_q (s)	n_α (g/rad)	DLC gain $\frac{K_{DLC}}{K_{DLC_{max}}}$	System parameters (roll) τ'_R (s)	T_p (s)
Long-short	$\boxed{\text{E-5}}$	1.31	0.62	0.96					
period	E-4	1.12	0.63	1.15					
frequency*)	E-3	0.94	0.66	1.37	0.07	4.71	0	0.23	0.02
	E-2	0.75	0.74	1.67					
	$\boxed{\text{E-1}}$	0.68	0.78	1.72					
Pitch	$\boxed{\text{E-5}}$	1.31	0.62	0.96					
rate	F-3	1.27	0.61	1.45	0.07	4.71	0	0.23	0.02
overshoot	F-2	1.24	0.62	2.56					
	$\boxed{\text{F-1}}$	1.23	0.63	5.26					
Maneuver	$\boxed{\text{G-1}}$	1.30		0.93			0		
enhancement	G-2	1.31	0.62	0.93	0.07	3.41	0.32	0.23	0.02
	$\boxed{\text{G-3}}$	1.32		0.92			0.68		
	G-4	1.33		0.92			1.00		
Roll mode	$\boxed{\text{T-1}}$							0.23	0.02
time constant	T-2	1.31	0.62	0.96	0.07	4.71	0	0.52	0.03
	T-3							0.86	0.04
	$\boxed{\text{T-4}}$							1.96	0.04
Roll time	$\boxed{\text{T-1}}$								0.02
delay	T-6	1.31	0.62	0.96	0.07	4.71	0	0.23	0.12
	$\boxed{\text{T-7}}$								0.22
	T-8								0.32

$\boxed{}$: Configuration evaluated in-flight.

*) The prefilter time constant has been varied such that $1/\tau_q \approx 0.8\ \omega_{n_q}$

AD-A118 596 ADVISORY GROUP FOR AEROSPACE RESEARCH AND DEVELOPMENT--ETC F/G 1/3
CRITERIA FOR HANDLING QUALITIES OF MILITARY AIRCRAFT.(U)
JUN 82
UNCLASSIFIED AGARD-CP-333 NL

2 of 4



HANDLING QUALITIES CRITERIA FOR LONGITUDINAL CONTROL

W. Neuhuber
Dornier GmbH
Postfach 1420
7990 Friedrichshafen
Germany

and

L. Diederich, K. Brauser
Messerschmitt-Bölkow-Blohm
Postfach 801160
8000 München 80
Germany

SUMMARY:

Current specifications (e.g. MIL-F-8785) are not fully applicable for augmented airplanes (high order systems). Major shortcomings are due to unsuitable mathematical representation of criteria. In addition current criteria are not precise enough, both for augmented and unaugmented systems.

Detrimental effects to flying qualities are discussed. Means to reduce these effects are proposed, including nonlinear methods.

With respect to the pitch short period a criterion (based on MIL-F-8785 and others) is presented, which states a more precise relationship between the relevant parameters.

When transformed into time history and frequency response, this criterion is applicable for any system.

Especially it is shown, that Nichols-plots can be a useful tool for handling qualities evaluation.

Optimal Stick Force Gradient Criteria ($F_s/n_{z/\alpha}$) are derived from a quasi-linear pilot-aircraft closed loop analysis.

The closed loop characteristics are formulated by means of a bandpass filter frequency response function, the characteristics of which are defined by the equivalent aircraft and a precision pilot model frequency function.

INDRODUCTION:

High performance fighter aircraft show extremely expanded flight regimes as measured by dynamic pressure and incidence angles. Additionally the static stability is reduced for reasons of improved performance.

These tendencies require the implementation of an automatic flight control system. Powerful FCS-computers allow sophisticated controls. Wide variations in control surface effectiveness and coupling effects require the adaptation of control surface blending, varying characteristics of the uncontrolled aircraft require the adaption of control laws parameters. Furthermore increased filtering of sensor signals is required, as high gain stabilization in an airframe with minimized structural weight poses problems associated with structural modes and their interaction with control system dynamics.

Altogether these measures end up in a high order dynamic system. The known MIL-criteria are no longer applicable without additional reasoning.

New and further improved criteria are needed to overcome deficiencies which could be noted in some early flight tests with augmented airplanes. The influence of detrimental effects have to be analyzed and taken into account at proper design stages. These analyses, together with improved criteria including these detrimental effects, will lead to better guidelines for FCS-design, hopefully.

Of primary interest are criteria with directly usable hints and design goals for the development of the flight control laws and the systems specification. The basic background of all flying qualities requirements is the question what should the handling and ride qualities of an aircraft be so that the pilot can fulfill his mission and flight task in a safe way with minimum workload, and within the limits of his ability to adapt to the control problems.

This formulation points out that the evaluation of these criteria are generally

influenced by effects sometimes considered marginal: density and update-rate of information presented to the pilot, e.g. visual cues on the head-up-display and general visibility, actuation of the pilot controls as stick, pedals, thumbwheels, discrete inputs etc.

1. Design Considerations for Longitudinal Control (by W. Neuhuber)

Deficiencies of FCS-designs encountered primarily in so-called tight control tasks as precise manoeuvering in air-to-air combat using the gun, landing approach and close formation flight or during inflight-refueling. In these cases the pilot shows high gain performing his task. In the above mentioned flight and mission phases dynamic pressure and angle of attack have strongly different values.

Therefore it can be expected that the definition and design of appropriate flying qualities will have to take nonlinear and time-varying effects into account.

Considering the response of the vertical acceleration n_z to pilot commands (Fig. 1-1) three different phases can be distinguished:

a) Initial lagged response
b) rise
c) settling to stationary value

In precise tracking tasks the initial phase (a) is of high importance. The time history is strongly influenced by the sum of time lags and delays located in the feed-forward loops. Additionally there is a nonminimum-phase effect with aircraft manoeuvering with horizontal tails or with vectored thrust alone. By blending manoeuver flaps with the actuation of these controls the nonminimum-phase effect can be eliminated. As a good "rule of thumb" the degree of elimination should be confined so that the initial pole of rotation lies at the pilot's station. In that case the pilot will sense a rotational but no translational acceleration at the first moment of aircraft response.

With regard to the allowable amount of time delay MIL-9490D gives stringent but justifiable limits which should be given notice early in the FCS design. Filtering in the feed-forward loop should be minimized for inherent time lags and delays.

In the phase (b) of the n_z-response (Fig. 1-1) the steepness of the n_z-build-up should not exceed some limit. Otherwise the pilot could be disconcerted primarily due to the effects of forces on his head and arms. A limiting value of 8 g per sec has been *cited /14/, but there should be a limit related to the steady state vertical accelerat*ion, e.g. from 1.6 to 3.6 times the steady state value in one second.

This rule was found by consideration of coordinated turns using full roll acceleration and trying to stay in almost constant altitude. Pilot comments on normal load buld-up experienced in highly manoeuverable aircraft will help to establish a more accurate limit.

In the phase (c) of the n_z-response (Fig. 1-1) a well damped behaviour is required. Settling time should be minimized. This requirement is in conflict with the need for quick rise time of n_z as far linear airframe dynamics and linear control laws are considered. For a short rise time in phase (b) a high frequency and a low damping of the short period motion of the aircraft would be favourable, but in phase (c) good damping is required. These conflicting requirements can be achieved simultaneously by means of nonlinear control laws with some gain adjustment depending on the difference between commanded and actual load factor.

Whereas in phase (a) time delays in the feed-forward loop were detrimental to performance, now time delays and lags in the feedback loops must be paid attention in order to satisfy the requirement of good damping especially in gusty environment.

The discrimination between three different phases of the load factor response (Fig. 1-1) finds its complement in the discrimination between gross manoeuvering and "fine tuning" of aircraft attitude. MIL-criteria pertaining to eigenvalue characteristics like frequency and damping are a good guideline to FCS design. This design should allow nonlinear control law modifications advised by analyses of mission-related manoeuvers. Conflicting requirements and aspects of the n_z-and q-response can be overcome by means of proper static and dynamic blending of aerodynamic control surfaces (and thrust vectoring), nonlinear shaping of gains in feedforward (command shaping filters) and feedback loops, and by inserting dynamically shaped pilot commands into different loops of the longitudinal control system. Unfortunately these means and measures are accompanied by detrimental effects like additional computer burden and therefore delay, so that a trade-off has to be accomplished between possible improvements and actual realizability.

2. Criteria for Pitch Short Period Motion (by L. Diederich, MBB Munich)

The method for derivation of short period handling qualities criteria proposed in this presentation is based on the assumption that essentially two parameters are relevant: the control anticipation parameter (CAP) and the overshoot ratio (resonance amplitude) /16/.

2.1 Derivation Of Optimum Parameters

The control anticipation parameter is defined as follows /15/, /17/:

$$CAP = \dot{q}_o/\Delta n_z = \dot{q}_o.g/q_{\infty}.V = T_\theta.\omega_n^2.g/V$$

(note: all parameters are to be understood as low order equivalents of high order systems).

With $(n_z/\alpha)_{equivalent} = V/g.T_\theta$ /17/, we obtain $CAP = \omega_n^2/(n_z/\alpha)_{equivalent}$, according to para 3.2.2.1.1 of reference /15/.

The resonance amplitude is defined as: $|A|\omega_n = T_\theta.\omega_n/2\cdot\zeta_n$

If an optimum value for $|A|\omega_n$ exists, then the optimum damping ratio must be variable:

$\zeta_n = T_\theta.\omega_n/2\cdot|A|\omega_n$

According to reference /11/ and results of references /6/ and /18/, the optimum amplitude is: $|A|\omega_n = 1.8 \hat{=} 5$ db. Thus the proposed replacement for para 3.2.2.1.2 of reference /15/ is as follows:

$\zeta_n = T_\theta.\omega_n/3.6$ and $\zeta_n\cdot\omega_n = T_\theta.\omega_n^2/3.6 = CAP.V/3.6g$

2.2 Proposed Tolerances

According to references /15/ and /17/, and results from /6/ and /18/, the relationship between pilot opinion rating (POR) and control anticipation parameter should be as follows: $POR \approx CAP$ for $CAP \geq 1$ and $POR \approx 1/CAP$ for $CAP \leq 1$.

Table 2-1 shows the proposed boundaries:

CONTROL ANTICIPATION PARAMETER

Level	Cat. A&C	Category B
1*	.5 to 2	.11 to 3.5
1	.29 to 3.5	.11 to 3.5
2	.15 to 6.5	.11 to 6.5
3	.11 to 9.5	.11 to 9.5

Level 1* = optimum region

The requirements for category C in table 2-1 are more restrictive than those in reference /15/. The reason is, that a high value of CAP will help to cure problems associated with the stick pumping phenomenon as mentioned in reference /17/ and as experienced with several systems: an improved pitch acceleration response due to an adequate value of CAP will reduce pilot's pumping activity (controler and surface amplitudes) during flare, and, on the other hand, short period and pumping frequencies will be well separated, so reducing the probability of pumping activity to end up in pilot induced oscillations.

Since deviations of damping ratio from its optimum value result in similar degradations of flying qualities as deviations of ω_n from optimum do, the tolerances of both parameters should be coupled as follows:

$\zeta_{max}/\zeta_{min} = CAP_{max}/CAP$ for $CAP \geq 1$

and $\zeta_{max}/\zeta_{min} = CAP/CAP_{min}$ for $CAP \leq 1$

with CAP_{max} and CAP_{min} according to table 2-1.

2.3 Development of Criteria

If one expresses mentioned requirements in different mathematical representations, one can obtain a set of equivalent short period criteria.

For example we assume an arbitrary condition as follows: V = 100 m/sec, $n_{z\alpha eq}$ =16g/rad, T_θ = 0.629 sec, which requires: ω_n = 4.03 rad/sec, ζ = 0.703.

Figure 2-1 shows the optimum of ω_n and ζ and the level 1 boundaries in comparison to those of reference /15/. Several parameter combinations tolerated in ref. /15/ are excluded by proposed requirements. Note that location of optimum and boundaries will shift with change of flight condition and configuration.

Figure 2-2 shows time response of pitch rate for control stick step input, with the parameters stated before.

Figure 2-3 shows the level 1 locations of poles in s-plane for mentioned example.

Figure 2-4 shows the corresponding Bode plots of pitch rate vs. control stick input. Note that the high frequency asymptotes depend on system order. Rather than the example of reference /11/, our one is low order. For frequencies higher than short period resonance rather the relationship between amplitude and phase than their absolute values should be relevant.

Therefore an alternative representation in frequency domain is proposed: a pseudo-Nichols diagram of pitch attitude vs. stick input according to figure 2-5.

Figure 2-6 shows the proposed boundaries as developed from the requirements mentioned before, and as checked against data set from references /6/ and /18/, and actual systems. This representation is not subject to changes due to variations of flight condition, configuration or system's order. However, bandwidth should be verified (BW ≈ ω_n).

Table 2-2 shows some properties of the criteria mentioned:

Representation	Required Actions	
	Find low order equivalent	adapt for flight condition
ω_n vs. ζ	yes	yes
time history	no	scaling of time axis
S-plane	yes	yes
Bode plots	no	scaling of ω-axis & high frequency asymptotes=F(order)
Nichols diagram	no	evaluate bandwidth: BW ≈ 0.5 ω_n to 2 ω_n

The representation according to figures 2-5 and 2-6 was dominated as "pseudo"-Nichols diagram, since the mentioned functions do not contain any pilot model; loop closure via pilot is fiction for this representation. Therefore this criterion should not be misunderstood as comparable with the Neal & Smith-criterion /6/ or with the representation of part 3. of this report. Nevertheless, the closed loop curve of Nichols diagram showed to be useful boundaries for the proposed criterion.

Regardless of advantages of disadvantages of the representations presented, it is recommended to use more than one criterion simultaneously. Their choice may depend on mathematical tools and data set available or preferred by the user.

2.4 Concluding Remark

If future experience should require revision of proposed optimum values of control anticipation parameter and resonance amplitude, the method presented will not be affected. Just the numerical values will be subject to revision.

3. The Derivation of Pitch Stick Force Gradient Criteria from other Handling Qualities Criteria (by K. Brauser, MBB, Munich).

3.1 Pilot-Aircraft Closed Loop Analysis

Stick force-gradient criteria (e.g. for the derivation of the F_s/nz -slope) should be developed theoretically by closed loop analysis in order to introduce all handling criteria which can influence the stick characteristics, or which are dependent on these. The closed loop must contain a quasilinear pilot model which should be as precise as possible; and which has been validated by experimental work, and also by theoretical analysis.

The aircraft model used in this analysis should be an "equivalent" model such as has been presented in chapter 2 of this report.

The pilot model introduced here is given by fig. 3-1 (from Bubb /1/) which is similar to that of Mc Ruer and Magdaleno /2/ and which has been experimentally validated in /1/, theoretically also by the author (/3/).

Since the analysis will be performed at the frequency transfer level, the pilot model is represented by its frequency characteristics.

$$(1)\quad F_p(s) = \frac{K_p(1 + T_A s)(1 + T_V s)\, e^{-\tau_p s}}{\bar{K}(1 + T_W s)(1 + T_m s + T_m^{*2} s^2)}$$

$T_m = 2\zeta_m/\omega_m$, $T_m^* = 1/\omega_m$ } damping and resonance of the man-manipulator system
T_W = neuromuscular lag time constant
K_p = pilot gain
$\bar{K}$ = neuromuscular force and joint sensor feedback gain
T_A, T_V = information input and processor lead time constant
s = $\sigma + j\omega$

The equivalent aircraft model is given by (pitch axis short period only):

$$(2)\quad F_a(s) = \frac{K_c K_\theta (1 + T_\theta s)\, e^{-\tau_a s}}{s(1 + T_a s + T_a^{*2} s^2)} = \theta/F_s$$

F_s = control stick input
K_c = feel force gain
K_θ = stick-force to elevator-deviation gain
T_θ = $1/L_\alpha$, the lift coefficient lead time constant
$T_a = 2\zeta_{nsp}/\omega_{nsp}$, $T_a^* = 1/\omega_{nsp}$ } short period damping and resonance

The closed loop characteristics are given by (fig.3-2)

$$(3)\quad F_{CL}(s) = \frac{F_p(s)F_a(s)}{1 + F_p(s)F_a(s)}$$, which has the form

$$(4)\quad F_{CL}(s) = \frac{K_p K_c K_\theta (1 + T_A s)(1 + T_V s)(1 + T_\theta s) e^{-\tau_e s}}{\bar{K}\, s(1 + T_W s)(1 + T_a s + T_a^{*2} s^2) + K_p K_c K_\theta (1 + T_A s)(1 + T_V s)(1 + T_\theta s) e^{-\tau_e s}}$$

and in which the pilot neuromuscular manipulator element has been neglected, since ω_m should be at least 3.5 times higher than ω_{nsp} (/3/).

Using the Padé- substitution

$$(5)\quad e^{-\tau_e s} = \frac{1 - T_e s}{1 + T_e s}, \qquad T_e = \tau_e/2$$,the delay term is replaced by

$$(6)\quad e^{-\tau_e s} = e^{-(\tau_p + \tau_a)s} = \frac{1 - T_e s}{1 + T_e s}$$

which gives the possibility of the formulation of a denominator polynom in eq. (4) which now can be evaluated numerically.

The solution of the problem represented by equation (4) is found to be a "bandpass-filter" like transfer function

(7) $$F_{CL}(s) = \frac{(1 + T_A s)(1 + T_V s)(1 + T_\theta s)(1 - T_e s)}{K^*(s)(1 + T_e s)\cdot(1 + [\bar{K}/(\bar{K}^* K_e) - T_e]s + [\bar{K}T_W/(\bar{K}^* K_e)]s^2)\cdot(1 + T_a s + T_a^{*2} s^2)}$$

$$K_e = K_p K_C K_\theta$$

$$K^*(s) = \frac{(1 + T_V s)(1 + T_\theta s)}{1 + T_a s + T_a^{*2} s^2}$$, the pilot's gain compensation of the internal aircraft model.

(8) $\overline{K^*}$ = c./K*(s)/, a chosen constant average amplitude of K*(s) over the interesting frequency range ($c \leq 1$).

The filter characteristic given by eq. (7) is chosen as a complete handling criterion of the system pilot-aircraft, in which all single handling criteria known up to date can be expressed explicitly depending on each other. (Complete derivation is given in /8/ or /9/). The equation (7) is the solution for <u>all</u> possible aircraft characteristics of the type of eq. (2), for it describes primarily the role of the human pilot which closes the control circuit. This is true also for all aircraft states of a given aircraft. Any numerical solution should be understood as a solution of a quasilinear control circuit specified by the momentary operational state. A change of the system state is adapted by the pilot by generating new lead and gain constants (see table 3-1 for examples of states).

The hypotheses on which this analysis is based are the following:

(1) The loop pilot-aircraft is always closed when the pilot holds his hand on the stick, even in the cases when he tries to introduce singletts or doubletts into the circuit.

(2) The pilot generates a lead term T_V in order to build up an "internal aircraft model" using also the aircraft given lead term T_θ:

$$(1 + T_V s)(1 + T_\theta s) = (1 + T_a s + T_a^{*2} s^2) \quad , \quad T_V T_\theta = T_a^{*2}$$

(3) The pilot tries to compensate this "model" if $T_V + T_\theta > T_a$ (for $\zeta_a = T_a/2T_a^* < 1$)

by the introduction of a frequency dependent gain factor K*(s) (see above), which is unity for $T_V + T_\theta = T_a \geq 1.0$.

(4) The bandpass filter transfer function has a constant bandwidth ω_g = 3.5 rad/second, the phase shift at that frequency should not be larger than 90°.

(5) The pilot adjusts K_e (by means of K_p), K*(s), T_V (also T_A if necessary), T_W and T_e in order to keep the bandpass-filter characteristics of a closed loop containing an arbitrary aircraft within the limits given in fig. 3-3.

(6) The adaptibility of the pilot is limited by

- The order of the generated lead ($\leq$ 2 for good handling)
- Total lead time ($T_V + T_A \leq$ 2 ... 3 seconds) - see /4/, /5/, /6/ -
- $T_e = \frac{c_p + c_A}{2}$, since $\frac{c_p + c_A}{2}$ must not exceed 0,4 seconds in order to avoid PIO-prone characteristics.
- K_p is limited for high values by small force discrimination of the internal feedback loop, and for low values by maximal muscular tension or - at a lower level - by comfort.
- K*(s) is limited by attention level and the characteristics of the guidance function or the disturbances of the system.

  It is believed that the pilot introduces a constant factor K* which is an average amplitude value of K*(s) over the interesting frequency range.
- ω_k must not be "frozen" by $\omega_m \rightarrow \omega_k$. ($\omega_k$ is defined by equ. (9))

3.2 Discussion of the Solution

The slope and the limits of the transfer function (eq. (7)) shown in fig. 3-3 have been evaluated from the "Nichols theorem" developed in part 2 of this report. The limits of fig. 3-3 indicate the area of <u>good</u> handling qualities. It can be demonstrated that only good handling qualities criteria represented by the aircraft which are discussed in part 2 enable the pilot to hold the filter characteristics within these limits, which also contain the Neal and Smith criterion (dotted lines). A more detailed discussion is given in /8/. There are some conclusions which should be discussed in detail in this report, however.

3.2.1 The Pilot "Strategic Control" Term

In eq. (7) the term $(1 + (\frac{\bar{K}}{\bar{K}^* K_e} - T_e)\, s + \frac{\bar{K}}{\bar{K}^* K_e} T_W s^2)$ may be said to be the "Strategic control term", because it contains all adaptive pilot activities. It can be demonstrated, that

(9) $\frac{\bar{K}}{\bar{K}^* K_e} T_W = \omega_k$

is the maximal stick movement frequency applied to solve control problems, while

(10) $\frac{\bar{K}}{\bar{K}^* K_e} - T_e$ is the "damping factor" $\frac{2\zeta_k}{\omega_k}$

of this term. If ζ_k is too low, it gives rise for PIO tendencies (see 3.2.2). A high value of ω_k is correlated with good handling qualities, this was explicitly proven by the experimental work of Miller /10/, in which was found that a smallest tracking error is correlated to a high value of ω_k ($2 \leq \omega_k \leq 6$ rad/sec). The numerical values in table 3-1 for stable solutions of eq. (7) show the same tendency.

The "strategic term" also contains the adaptive gain factors $K_e = K_p K_c K_e$ which the pilot can adapt to stabilization or tracking problems by variation of K_p, or $K^*(s)$ which he applies to build up the "internal aircraft model" as a feedback function, and last not least $\bar{K}^*$ which is the more realistic tool to compensate the aircraft's instabilities (see 3.2.3). These variable gain factors which are depending on the aircraft momentary state are used by the analyst to derive optimal stick force characteristics for a given aircraft characteristic (see 3.3).

3.2.2 Criteria for Pilot Induced Oscillation (PIO)

There are several possible sources of PIO, which are delay and low damping. If delay terms are present and exceed a certain limit (e.g. $\tau_e > 0{,}4$ sec) the damping of the "Strategic term"

(10a) $\zeta_k = (\frac{\bar{K}}{\bar{K}^* K_e} - T_e) \cdot \frac{\omega_k}{2}$

will decrease, this corresponds to a smaller K_e or higher feel spring constant to hold the slope of eq. (7) within the "good" boundaries. Also the pilot is able to decrease τ_e to a certain amount since $\tau_e = \tau_p + \tau_a$, and at least a certain amount is compensated by a second lead term, say $1 + T_A$ S. But, delay compensation by lead is known to be never satisfactory. The rest of $\tau_e = \tau_p + \tau_a$, which is a system given delay τ_a, will cause the filter function slope (fig. 3) to exceed the given limits, if its value is too high. As a result τ_a increases the PIO tendencies if its value exceeds 0,3 sec.

Since ζ_k is one criterion for PIO tendency, also the factor ω_k of eq. (10a) can give rise to PIO. ω_k is dependent on $\bar{K}^*$, K_e, and T_W, the neuromuscular man-manipulator lag. This lag should be - according to Mc Ruer /2/ - as low as 0,1 seconds or smaller, but an adaption of T_W by the pilot means effort and therefore decreases his handling quality rating.

Another source of PIO (high frequency bobble) is low damping ζ_a of the aircraft in higher energy states. If the pilot must adapt to it by means of low gain $K^*(s)$, again the slope of the filter function will exceed the "good" limits of fig. 3. $K^*(s)$ corresponds directly to the "droop" phenomenon described by Neal and Smith /6/, and is discussed in the following chapter.

3.2.3 "Droop" Phenomena

The "droop" described in /6/ depends on the pilot's ability to compensate the aircraft's dynamics by lead, lag, or gain. The result of this analysis is that the gain factor $K^*(s)$ - from equation (7) - which is a feedback transfer function necessary for the exact numerical solution of the problem eq. (4) cannot be aligned by the pilot himself exactly because the frequency slope of $K^*(s)$ shown by fig. 3-4 requires too much attention and knowledge of the aircraft and environment state, and is sometimes blocked by the manipulator system characteristics. Therefore the pilot will replace the frequency slope by a constant average amplitude $\bar{K}^*$ of $K^*(s)$ over the interesting frequency range. This factor $\bar{K}^*$ automatically gives rise for a "droop" response instead of the ideally flat filter response which is only true for $\zeta_a \geq 1{,}0$ in real cases.

3.2.4 Evaluation of an Example

Table 3-1 contains a data sample of closed loop characteristics for four States of a chosen aircraft. The four states are indicated by the equivalent aircraft characteristics T_θ, $N_{z\alpha}$ [1]) V, T_a and $T_a{}^*$ which were chosen to meet the "CAP"-requirements of MIL-F-8785B. T_e was set to 0,1 seconds, and the pilot variables T_A, T_V, $\bar{K}$, K_e and $K^*(s)$ resp. $\bar{K}^*$ were set for each of the four states in order to keep the resulting filter response within the limits. This response is shown by fig. 3-5, which demonstrates the "droop" characteristic using $\bar{K}^*$, and phase and amplitude shifts according to T_e.

This example now is analysed in order to extract the force-gradient requirements from the variable gain terms $\bar{K}/\bar{K}^*K_e$.

3.2.5 Manoeuvering

Manoeuver inputs of the pilot are - according to a widespread opinion - actions which open the closed circuit. Actually this may be not true. If the hypothesis (1) can be applied also to hard manoeuvre inputs - and without doubt the pilot has to keep his hand on the stick in such moments - the system is also closed. In /8/ it is shown that according to fig. 3-2 another feedback mechanism is switched on, or the pilot gain is now K_p instead of K_p/s. This means: feedback of $\dot{\theta}$ instead of θ (see also /7/). After /8/ the closed loop transfer function now can be written as

$$\text{(7a)}\quad F_{CL}(s) = \frac{(1 + T_A s)(1 + T_V s)(1 + T_\theta s)e^{-\tau_e^* s}}{K^*(s)(1 + \bar{K}/(\bar{K}^* K_e))\cdot(1 + T_W s)(1 + T_a s + T_a^{*2} s^2)}\ ,\quad \tau_e^* \leq \tau_e$$

which is almost identical with the open loop. But, here the gain of the circuit has changed from $\frac{Ke}{\bar{K}}$ to $\frac{1}{K^*(s)\ (1 + \frac{\bar{K}}{\bar{K}^*\ K_e})}$, and again $\frac{1}{K^*(s)}$ is the source of "droop" phenomena. But, in this case, the pilot has to compensate his neuromuscular lag $1 + T_W s$ by his lead term $1 + T_A S$, if this is necessary, while τ_e is decreasing below τ_e.

3.3 The Derivation of the Stick Force Gradient F_s/nz from Validated $\frac{\bar{K}}{\bar{K}^*\ K_e}$

The hypothesis is that $\bar{K}/\bar{K}^*K_e$ is proportional to F_s/nz, and should be adjusted by the pilot in order to keep the system response within the stability limits.
We have

(11a) a) $\bar{K} = 1 + K_k + K_p K_w$ (see /3/, also fig. 3-1)

b) $K_k = \frac{Fs/nz}{F_s\ max}$ $\left[\frac{N}{N}\right]$ = gain factor of force sensor feedback

c) $K_w = \frac{\delta}{\delta max}$ $\left[\frac{rad}{rad}\right]$ = gain factor of joint sensor feedback

d) $K_p = \frac{F_s\ max}{F_s/n_z}$ $\left[\frac{N}{N}\right]$ = pilot force input gain

e) $K_e = K_p K_C K_\theta K_V K_A$

f) $K_C = \frac{F_s}{\delta}$ $\left[\frac{N}{rad}\right]$ = feel force gain

g) $K_V = \frac{\Delta\theta}{\Delta\theta_s}$ $\left[\frac{rad}{rad}\right]$ = tracking error threshold gain ($\Delta\theta_s$ = threshold angle θ)

h) K_A = is set one = (e.g.: display gain)

i) $K_\theta = \frac{\dot{\theta}}{F_s}$ $\left[\frac{rad}{Nsec}\right]$ = elevator gain

j) $\bar{K}^*$ = constant average amplitude factor, which depends mainly on ζ_a, and — , and which is calculated from eq. (8) $\bar{K}^* = C/K^*(\omega = \omega_a)/$, $C \leq 1{,}0$

[1]) $N_{z\alpha}$ according to ch. 2 to replace n_z/α

Assumptions:

$F_{s\,max}$ = 146 N (= 32 lb) from MIL-F-8785 B
δ_{max} = 30° = 0,523 rad
$\Delta\theta_s$ = 0,052 rad (see /8/)

$$= \frac{\dot{\theta}}{F_s} \cdot \frac{F_s}{n_z} \cdot t_v,\ t_v = 1 \text{ sec.}$$

$$K_V = \frac{F_s}{n_z(=1)} \cdot \frac{\dot{\theta}}{F_s} \cdot \frac{t_v(=1)}{\Delta\theta_s}$$

All interesting values are calculated from eq. (11a - j) for the four states (ω_a = 1, 2, 3.5 and 7 rad/sec) and inserted into table 3-2 and compared with the values of $\frac{K}{K^* K_e}$ found in the analysis.

The result is that the values F_s/n_z calculated from eq. (11a - j) have a trace in fig. 3-6 which is in accordance with MIL-F8785B recommendations, the slope of which is not linear.

Another example which has been described in /8/ is shown by fig. 3-7. This example has T_θ-, and ω_a-values which are nearer to CAP = 1,0, and for which $F_{CL}(s)$ satisfies also the requirements of fig. 3. Aircraft characteristics of this example have been selected by means of the Northrop-requirements published by Gallagher and Nelson /11/. The F_s/n_z-range of the Northrop criteria are shown in both figures 3-6 and 3-7 as an optimal area.

3.4 Conclusions

It has been demonstrated that closed loop analysis including a physiological proven pilot model has the power of exact prediction of F_s/n_z-values by use of the bandpass filter criteria.

A comparison of this method with similar methods published by other authors (e.g. R. Hess /12/, /13/) showed that the application of an appropriate pilot model is very important and gives rise to more new handling criteria with predictive power. As it was proven in /13/, the closed loop analysis is the area of more insights into the various handling quality criteria depending on each other. Manoeuver handling criteria may be based on open loop analysis further, or, as was demonstrated here on quasi-open loop analysis which was performed effectively in the past.

Other stick requirements are also implicitly present in this analysis. The F_s/n_z-values have been calculated in para. 3.3 for the heavy centerstick only. In fact, "ministick" configurations will be preferred for future fighter aircraft (center or side location) which is indicated by the comments on eq. (4) dealing with the requirement

$$\omega_m \geq 3{,}5\ \omega_{nsp}$$

This means that the undamped resonance frequency of the man-manipulator system

$$\omega_m = 1/\sqrt{\frac{C_m}{K_m}}$$

should be that of a light-weight stick with high feel spring constant C_m, engaged preferably by the hand only while the arm is at rest. Values of $F_{s\,max}$ as are required for the calculations (11a ... 11j) need careful investigations before.

3.5 References

/1/ Bubb, P.: Untersuchung über den Einfluß stochastischer Rollschwingungen auf die Steuerleistung des Menschen bei Regelstrecken unterschiedlichen Ordnungsgrades (in German)
Dissertation TU München (1978)

/2/ McRuer D.T., Magdaleno R.E.: Human Pilot Dynamics with Various Manipulators
AFFDL-TR-66-138 (1966)

/3/ Brauser, K.: Untersuchung über ein dynamisches Modell des Zusammenwirkens von Pilot und Bedienelement mit Anwendung auf die Steuerbarkeit (in German).
MBB-TN-FE301-06/80 (1980). Also:

Theoretical Linear Approach to the Combined Man-Manipulator System in Manual Control of an Aircraft
MBB-M/FE301/S/PUB/43 (1981)
Report held on the 17th Annual Conf. on Manual Control, Los Angeles, 1981.

/4/ Anderson, R.O.: Theoretical Pilot Rating Predictions, in: AGARD CP-106, 1971

/5/ Arnold, I.D.: An Improved Method of Predicting Aircraft Longitudinal Handling Qualities Based on the Minimum Pilot Rating Concept. Masters Thesis No. GGC/MA/73-1 AFFDS, Wright Patterson AFB/Ohio, June 1973

/6/ Neal T.O., Smith R.E.: An Inflight Investigation to Develop Control System Design Criteria for Fighter Airplanes. AFFDL-TR-70-74, Vol. I and II (1970)

/7/ Hess, R.A.: Dual Loop Model of the Human Controller J. of Guidance and Control, Vol. 1, No. 4, 1978

/8/ Brauser, K.: Die Ableitung von Steuerbarkeitskriterien für Kampfflugzeuge aus einem quasilinearen Präzisionsmodell des Menschen mit Hilfe der Filtertheorie (in German). MBB-FE301/S/STY/33 (1981)

/9/ Brauser, K.: The Derivation of Handling Quality Criteria from Precision Pilot Model Characteristics. Paper submitted to the IFAC/IFIP/IFORS/IEA Conference on Analysis, Design and Evaluations of Man-Machine-Systems Baden-Baden, 1982

/10/ Miller, U., Schwegler, F., Schmidt K.: Anthropotechnische Untersuchungen zur Pilotenbelastung und -Leistung. Teil 2: Untersuchung zur Tracking- und Wahrnehmungsleistung (in German). MBB-UFE-799/2-71 (1971)

/11/ Gallagher J.T., Nelson W.E.: Flying Qualities Design of the Northrop YF-17 Fighter Prototypes. Society of Automotive Engineer Business Aircraft Meeting, Wichita, 1977.

/12/ Hess, R.A.: A pilot modelling Technique for Handling Quality Research, Report to appear in: Journal of Cybernetics and Information Sciences (by personnel communication).

/13/ Hess, R.H.: A structural Model of the Adaptive Human Pilot AIAA paper 79-1784 (1979)

/14/ Guthrie, C.H.:Evaluation of Direct Force Mode Fighters by Combat Simulation. Report,submitted to the AGARD -FMP Symposion, Florence, Italy, 1981.

/15/ Anonymous: Military Specification: Flying Qualities of Piloted Airplanes MIL-F-8785 B (ASG) 1969

/16/ Diederich, L.: Gütekriterien für die Anstellwinkelschwingung MBB-TN-FE132-1/80, 1980

/17/ Bihrle jr., W.: A Handling Quality Theory for Precise Flight-Path Control. AFFDL-TR 65-198, June 1966

/18/ Smith, R.E.: Effects of Control-System Dynamics on Fighter Approach and Landing Longitudinal Flying Qualities, Vol. I AFFDL-TR 78-122, March 1978

aircraft state characteristics						Pilot state characteristics									Circuit characteristics		Comment	
V m/sec	n_z/α g/rad	[illegible]	ζ_a	T_θ sec	T_e sec	T_A sec	T_V sec	T_W sec	$\frac{\bar{K}}{K^* K_e}$ -Te	$\frac{\bar{K}}{K^* K_e}$ TW	ω_k rad/sec	ζ_k	$\frac{\bar{K}}{K^* K_e}$ sec	$\bar{K}^*$ ($K^*(s)$)	ω_g rad/sec	$\Delta\phi$ at ω_g	CAP fig.	Slope 3-5
50	3,8	1	0,7	1,4	0,1	0,1	0,7	0,1	0,455	0,0555	4,24	0,97	0,555	1,225	3,5	- 96	0,25	o
50	3,0	1	0,5	1,7	0,1	0,1	0,4	0,1	0,34	0,044	4,76	0,81	0,44	1,449	3,5	- 95	0,3	
50	3,8	1	0,7	1,4	0,3	0,3	0,7	0,1	0,58	0,088	3,37	0,97	0,88	1,225	3,5	-139	0,25	●
100	5,5	2	0,7	0,9	0,1	0,1	0,27	0,1	0,41	0,051	4,42	0,91	0,51	1,288	3,5	- 98	0,3	□
100	5,5	2	0,7	0,9	0,2	0,2	0,27	0,1	0,59	0,069	3,8	0,93	0,69	1,288	3,5	-124	0,3	
100	5,5	2	0,7	0,9	0,1	0,1	0,27	0,1	0,41	0,051	4,42	0,91	0,51	1,288	3,5	-108	0,3	
200	33,0	3,5	0,7	0,6	0,1	0,1	0,136	0,1	0,42	0,052	4,38	0,92	0,52	1,356	3,5	- 99	0,3	
	26,0	3,5	0,7	0,8	0,1	0,1	0,102	0,1	0,42	0,052	4,38	0,92	0,52	1,496	3,5	- 95	0,5	
	26,0	3,5	0,7	0,7	0,1	0,1	0,102	0,1	0,40	0,05	4,47	0,89	0,5	1,496	3,5	- 95	0,5	▲
400	100,0	7,0	0,8	0,4	0,1	0,1	0,051	0,08	0,405	0,042	4,87	0,98	0,505	1,778	3,5	- 76	0,4	+
	100,0	7,0	0,5	0,4	0,1	0,1	0,051	0,08	0,405	0,042	4,87	0,98	0,505	1,778	3,5	- 81	0,4	*

Table 3-1 Calculated data for four states of an equivalent aircraft (see frequency responses of fig. 5)

Characteristics Symbol	Dimens.	a/c state values (T_e = 0,1 sec - C_e = 0,2 sec)					Comments
ω_a	$\frac{1}{sec}$	1	2	3,5	7,0		
T_θ	sec	1,7	0,9	0,8	0,4		
V	$\frac{m}{sec}$	50	100	200	400		
$K_\theta = \frac{n_z}{F_s} \cdot \frac{g}{V} = \frac{\dot{\theta}}{F_s}$	$\frac{rad}{Nsec}$	0,00196	0,001886	0,00175	0,00136		
$K_V = F_s \cdot \frac{\dot{\theta}}{F_s} \cdot \frac{1}{\Delta\dot{\theta}_s}$	$\frac{rad}{rad}$	3,769	1,886	0,942	0,472		K_V may be larger
$K_p = \frac{F_s\ max}{F_s/n_z}$	$\frac{N}{N}$	1,46	2,81	5,214	8,111		
$K_C = \frac{F_s\ max}{\delta\ max} \cdot 57.35$	$\frac{N}{rad}$	280	280	280	280		
$K_e = K_p K_V K_C K_0$	$\frac{1}{sec}$	3,019	2,79	2,406	1,45		
$K_K = \frac{F_s/n_z}{F_s\ max}$	$\frac{N}{N}$	0,684	0,356	0,192	0,123		
$K_W = \frac{\delta}{\delta\ max}$	$\frac{rad}{rad}$	0,684	0,356	0,192	0,123		
$\bar{K} = 1+K_k+K_p K_W$		2,682	2,356	2,192	2,1233		
$\bar{K}^*$		1,457 to 2,1	1,288 to 1,67	1,449 to 2,255	1,778 to 3,17		Max $\bar{K}^*$ is to be considered
$\frac{\bar{K}}{K_e \bar{K}^*}$	sec	0,61 or 0,444	0,655 or 0,505	0,628 or 0,409	0,823 or 0,462		lower value according to max. $\bar{K}^*$
$\frac{\bar{K}}{K_e \bar{K}^*}$	sec	0,44	0,51	0,52	0,505		average from table 3-1
Fs/nz	$\frac{N}{g}$	100	52	28	18		

$F_s\ max$ = 146 N = 32 lb (within MIL recomm.)
$\delta\ max$ = 30° = 0,523 rad

Table 3-2: Values of Fs/nz calculated from the values of $\frac{\bar{K}}{K^* K_e}$ for the example aircraft in ch. 3.2.

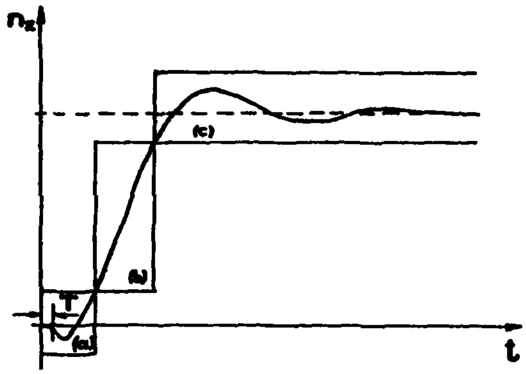


Fig. 1 - 1: Intersection of load factor response.

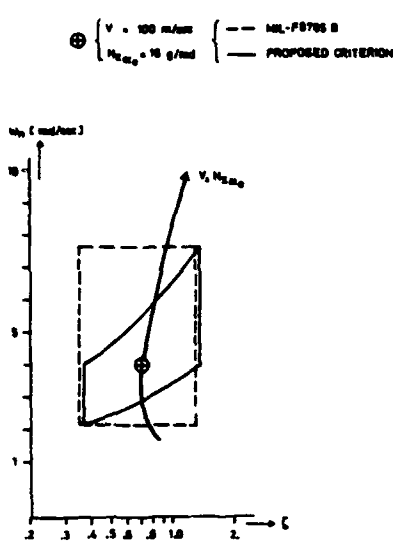


Fig. 2 - 1:Level - 1 boundaries for ζ and ω_n.

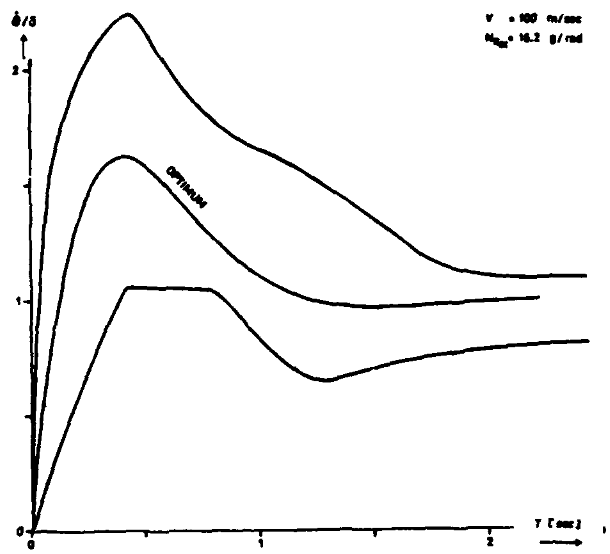


Fig. 2 - 2:Time response for level 1.

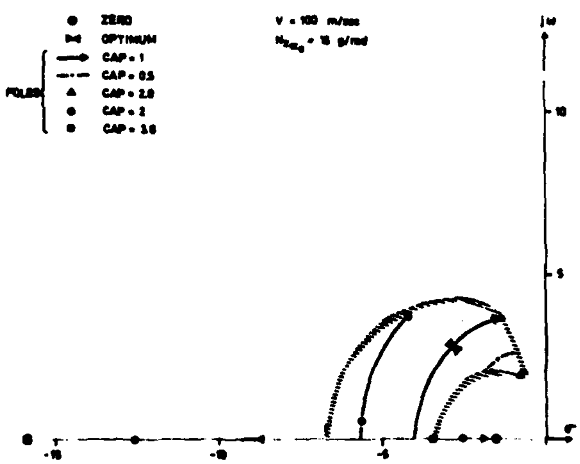


Fig. 2 - 3:Level -1 -area in the s -plane.

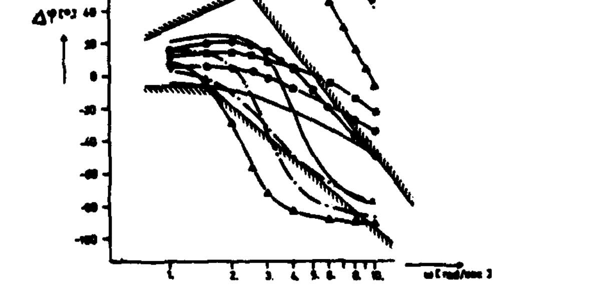


Fig. 2 - 4:Frequency response of pitch rate vs. stick input.

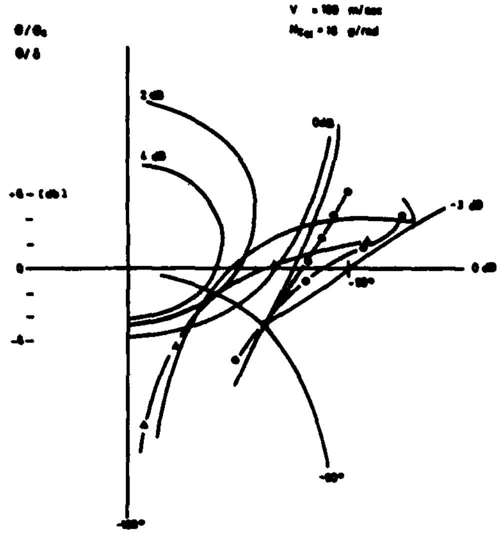


Fig. 2 - 5:Proposed short-period criterion.

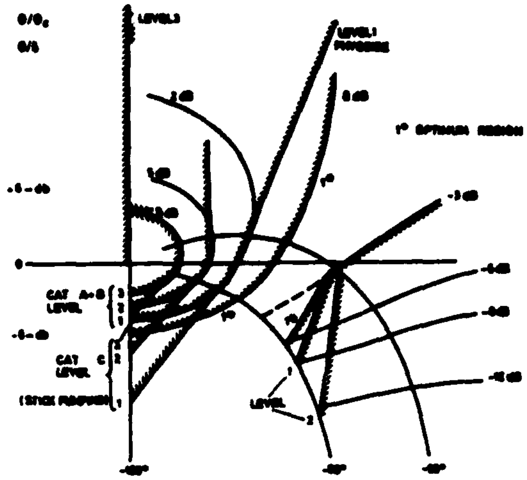


Fig. 2 - 6:Nichols plots of level -1 conditions.

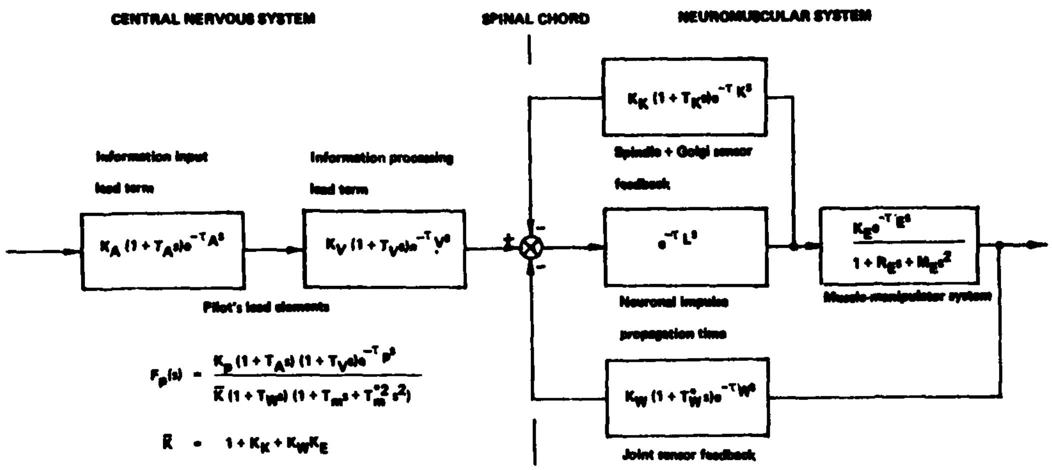


Fig. 3 - 1: Quasilinear precision model of the human controller (P.Bubb,/1/).

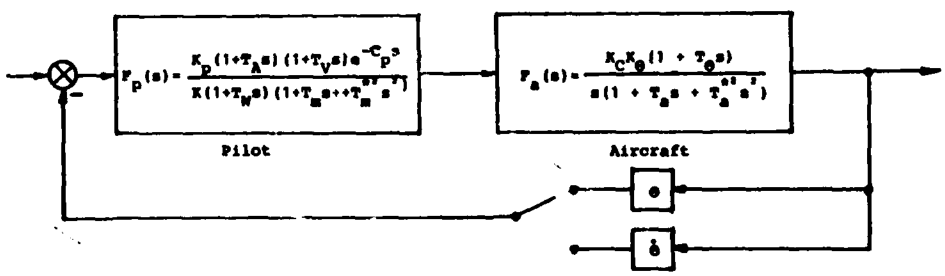


Fig. 3 - 2: Closed loop of the pilot-aircraft system with "dual control mode" feedback.

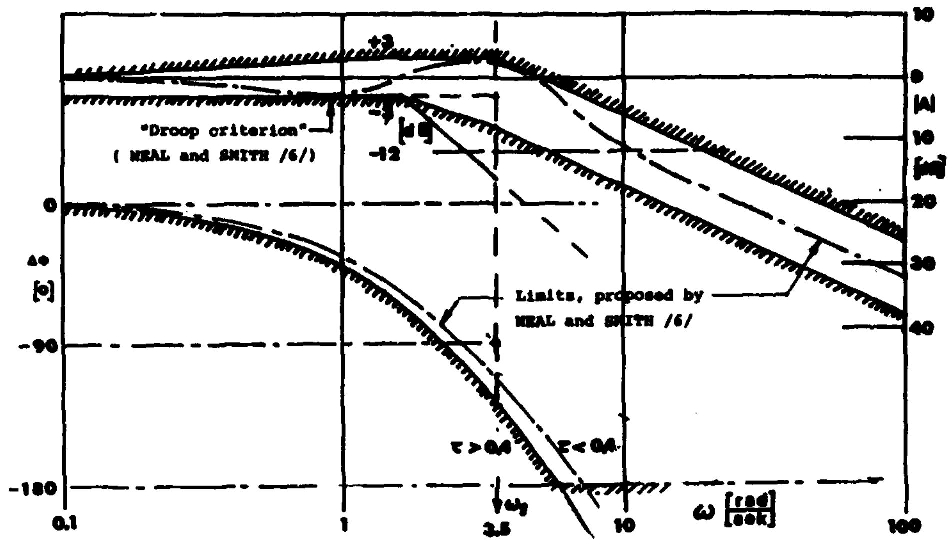


Fig. 3-3:The closed-loop handling qualities criterion.Limits for "good" handling: Criterion of Neal and Smith with "droop":

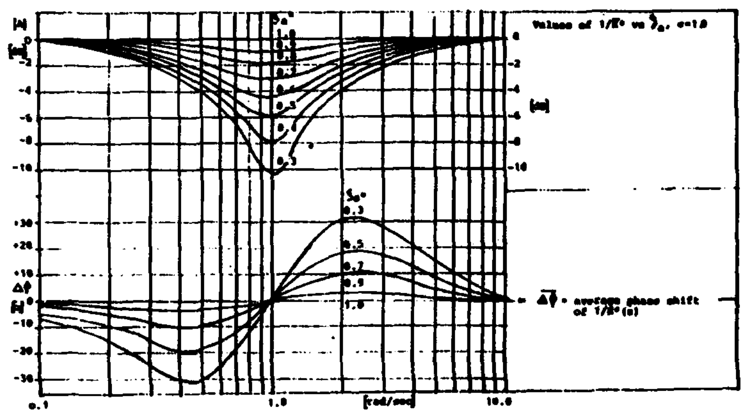


Fig. 3-4 : Frequency response for $1/K^*(s) = \dfrac{1 + T_a s + T_a^{*2} s^2}{(1 + T_V s)(1 + T_\theta s)}$, ζ_a = variable

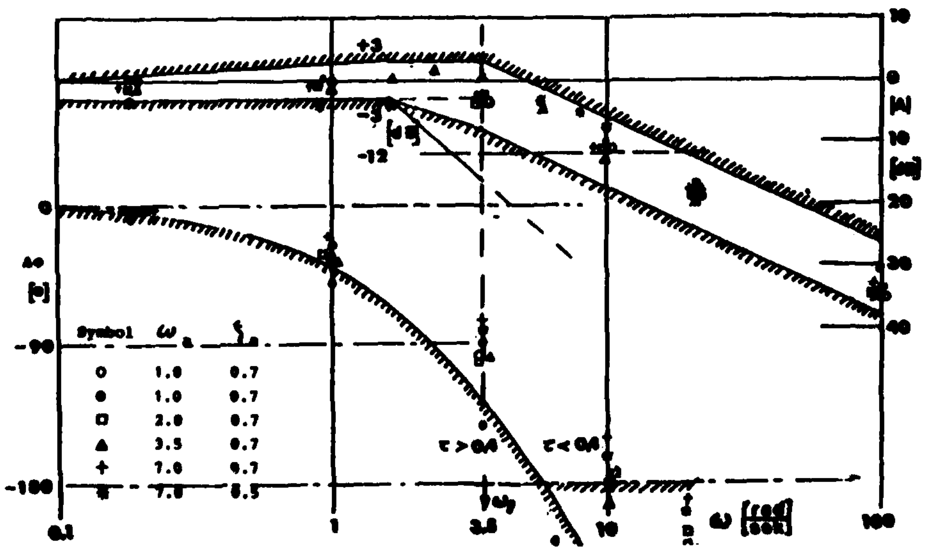


Fig. 3-5 : Bode diagram of $F_{CL}(s)$ for 4 aircraft states (flight conditions) ,data used from table 3-1. Droop is shown for the states ▲ and # .The state • exceeds the phase limits.

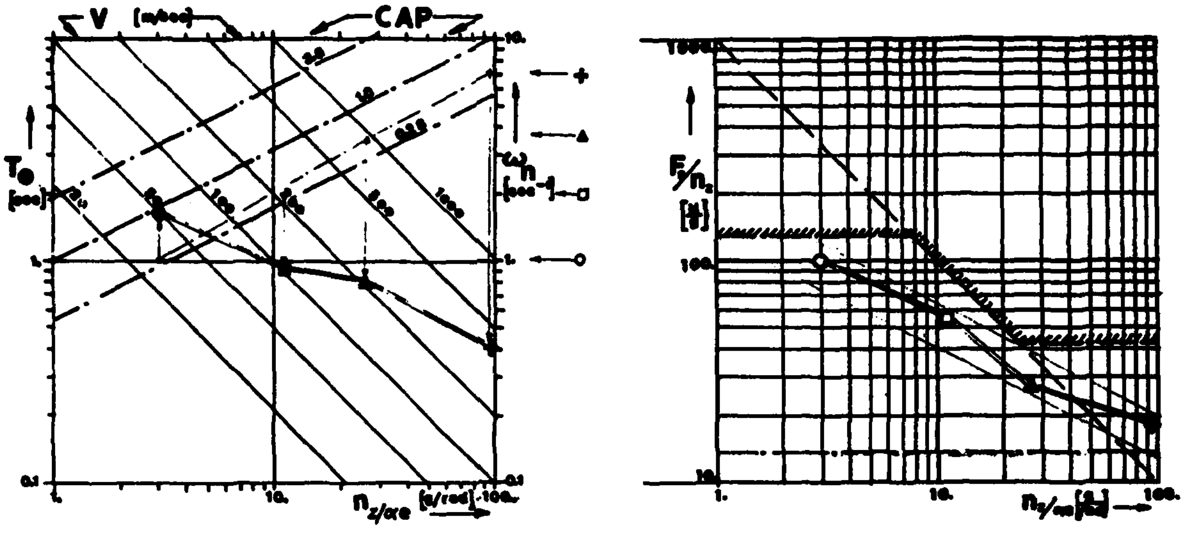


Fig. 3-6 : a) Aircraft state parameter T_θ vs. velocity V, $n_{z/\alpha}$, and ω_{nsp}.
The Control Anticipation Parameter is calculated from CAP $= \omega_{nsp}^2/n_{z/\alpha}$.
b) Aircraft state values of F_s/n_z calculated from $\bar{K}/(\bar{K}*K_e)$
The symbols are according to table 3-1.

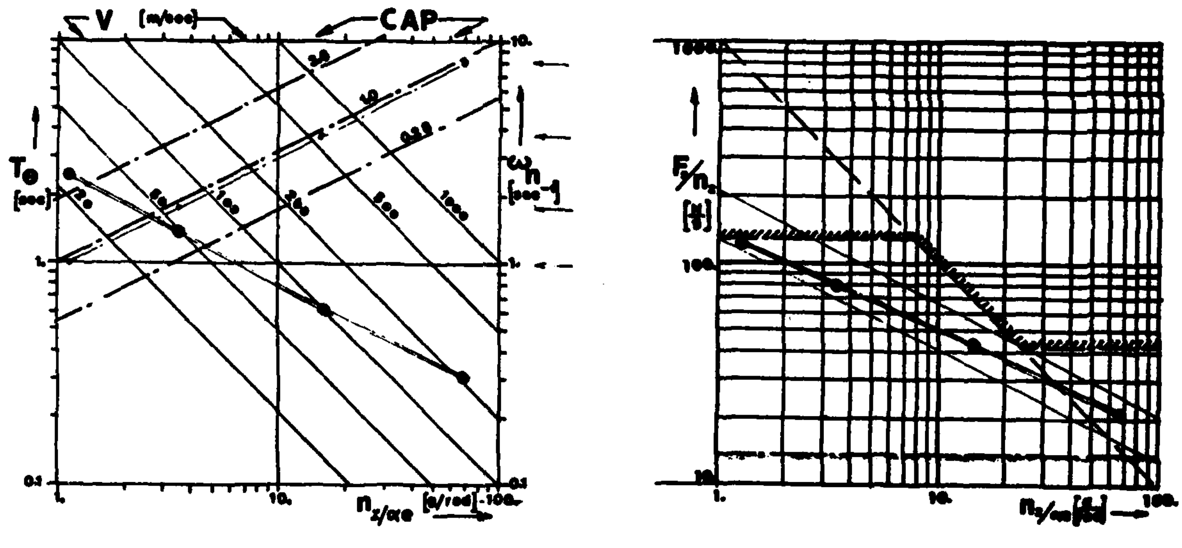


Fig. 3-7 : Aircraft state parameters and state values of F_s/n_z from another example /8/.

BANDWIDTH — A CRITERION FOR HIGHLY AUGMENTED AIRPLANES

Roger H. Hoh, Principal Research Engineer
David G. Mitchell, Staff Engineer, Research
Systems Technology, Inc.
13766 S. Hawthorne Blvd.
Hawthorne, CA 90250

John Hodgkinson, Unit Chief, Technology
McDonnell Aircraft Co.
P. O. Box 516
St. Louis, MO 63166

ABSTRACT

A criterion to discriminate between desirable, acceptable, and unacceptable handling qualities for highly augmented airplanes is presented. The criterion is based on an old and well accepted idea; namely, that bandwidth is a key measure of the quality of an airplane's handling characteristics in a tight tracking situation. Correlations are made using recent experimental data for pitch attitude control. Possible shortcomings of the criterion are also discussed.

INTRODUCTION

The criterion presented in this paper originated from an old and well accepted idea. Namely, that a measure of the handling qualities of an airplane is its response characteristics when operated in a closed loop compensatory tracking task. The maximum frequency at which such closed loop tracking can take place without threatening stability is referred to as "bandwidth" (ω_{BW}). It follows that airplanes capable of operating at a large value of bandwidth will have superior performance when regulating against disturbances.

When flying an aircraft with low bandwidth, the pilot finds that attempts to rapidly minimize tracking errors result in unwanted oscillations. He is, therefore, forced to "back off" and accept somewhat less performance (larger and more sustained tracking errors). It is not difficult to imagine a clear cut preference on the part of pilots for aircraft with increased bandwidth capabilities. In this paper, a quantitative definition of bandwidth is formulated and a handling quality criterion, correlated with a relatively large data base, is proposed.

BACKGROUND

As mentioned above, the concept of using bandwidth is not new. The most recent utilization of bandwidth was in the Neal-Smith criterion (see Ref. 1). This criterion consists of a grid of the closed loop pitch attitude resonance $|\theta/\theta_c|_{max}$ vs. pilot equalization for a piloted closure designed to achieve a _specified_ bandwidth. Experience with this criterion has shown that the results can be sensitive to the selected value of closed loop bandwidth. The criterion suggested in this paper utilizes the maximum value of bandwidth achievable without threatening stability, thereby removing the necessity for selecting a value for ω_{BW} a priori.

Another criterion utilizing bandwidth was suggested in Ref. 2. This criterion also selected a fixed value of bandwidth (1 rad/sec for power approach). It utilized the phase margin ϕ_M and slope of the phase curve $d\phi/d\omega$ at the selected bandwidth frequency as a correlating parameter. Again, experience has shown that the fixed value of bandwidth limited application of the criterion.

Most, if not all, familiar handling quality metrics are, in fact, a measure of bandwidth. However, these metrics tend to apply for classical airplanes which can be characterized by lower order systems. For example, the short term pitch response of a classical airplane is well represented by the familiar approximation (see Ref. 3)

$$\frac{\theta}{\delta_e} \doteq \frac{M_{\delta_e}\left(s + \frac{1}{T_{\theta_2}}\right)}{s\left(s^2 + 2\zeta_{sp}\omega_{sp}s + \omega_{sp}^2\right)} \tag{1}$$

It is easily shown for this (and similar) transfer function(s) that the quality of closed loop error regulation depends on the pilot's ability to increase the short period root (ω_{sp}) without driving it into the right half (unstable) plane. As illustrated by the generic sketches in Fig. 1, aircraft with low short period damping (ζ_{sp}) and/or low short period frequencies (ω_{sp}) tend to become unstable at low values of frequency (compare Fig. 1a and 1b).

Consider the bandwidth frequency, ω_{BW}, as occurring at some (for now) arbitrary margin below the frequency of instability (see Fig. 1). It can be seen from Fig. 1 that ω_{BW} depends uniquely on ω_{sp}, ζ_{sp} and $1/T_{\theta_2}$. Hence, these familiar flying quality metrics are, in fact, a measure of bandwidth. Again we make the point that bandwidth is _not_ a new idea.

The present impetus for using ω_{BW} as a criterion evolved from attempts to develop a flying quality specification for aircraft utilizing unconventional response modes with direct force controls (wings level turns, pitch pointing, etc.) (Ref. 4). The infinite variety of responses which could occur due to coupling within and between axes made it necessary for us to retreat to a more fundamental metric, which turned out to be bandwidth.

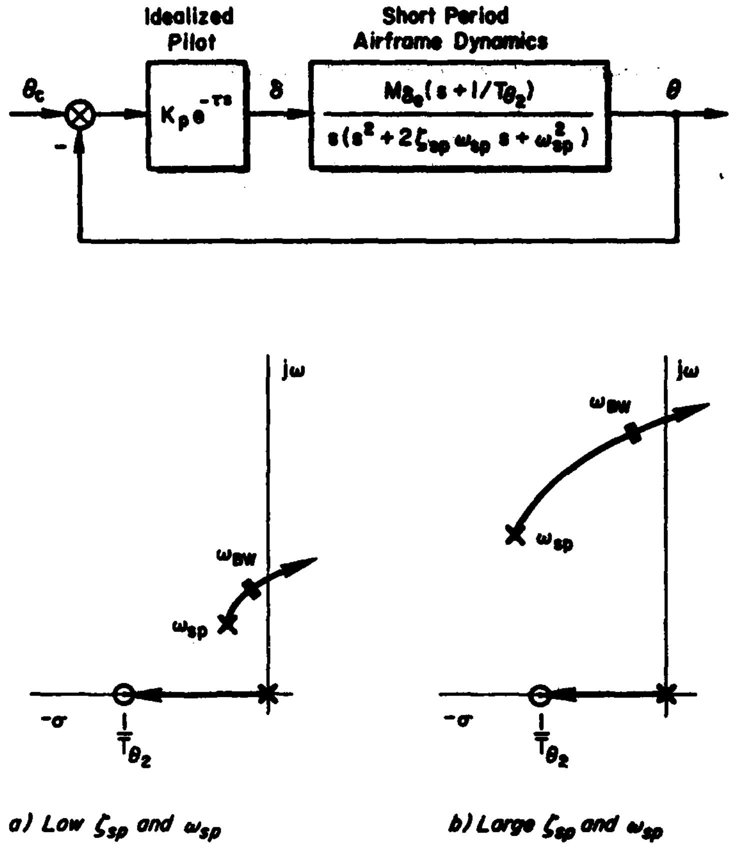


Figure 1. Simplified Pilot Vehicle Closure for Pitch Control

DEFINITION OF BANDWIDTH

The bandwidth (ω_{BW}) as defined for handling quality criterion purposes is the frequency at which the phase margin is 45 deg or the gain margin is 6 dB, whichever frequency is lower (Fig. 2). In order to apply this definition, one first determines the frequency for neutral stability from the phase portion of the Bode plot (ω_{180}). The next step is to note the frequency at which the phase margin is 45 deg. This is the bandwidth frequency as defined by phase, $\omega_{BW_{phase}}$. Finally, note the amplitude corresponding to ω_{180} and add 6 dB. Find the frequency at which this value occurs on the amplitude curve; call it $\omega_{BW_{gain}}$. The bandwidth, ω_{BW}, is the lesser of $\omega_{BW_{phase}}$ and $\omega_{BW_{gain}}$. If $\omega_{BW} = \omega_{BW_{phase}}$, the system is said to be phase margin limited. On the other hand, if $\omega_{BW} = \omega_{BW_{gain}}$ the system is gain margin limited; that is, the aircraft is driven to neutral stability when the pilot increases his gain by 6 dB (a factor of 2). Gain margin limited aircraft may have a great deal of phase margin ϕ_M, but increasing the gain slightly causes ϕ_M to decrease rapidly. Such systems are characterized by frequency response amplitude plots which are flat, combined with phase plots which roll off rapidly, such as shown in Fig. 2.

DATA CORRELATIONS

Several sets of data were correlated with bandwidth using the above definition. A typical result is shown in Fig. 3 utilizing the data from Ref. 1. While there is a definite pilot rating trend with ω_{BW}, the scatter for bandwidths between 2 and 6 rad/sec does not allow quantitative definitions of flying quality levels. A detailed analysis of the pilot/vehicle closure characteristics was made for configurations 1D and 2I. This was done to determine why these two configurations with nearly equal ω_{BW} would have such a large difference in pilot ratings (4 and 8 respectively). The detailed pilot vehicle closures are shown in Figs. 4a and 4b. The value of bandwidth is seen to be about the same for both cases. However, if the pilot were to track very aggressively by further increasing his gain (to operate at frequencies above ω_{BW}) Configuration 1D would only be unstable for very high pilot gains whereas 2I would rapidly become unstable (compare the root loci in Figs. 4a and 4b). This behavior is predictable from the phase curves. In particular, Configuration 1D has a phase curve which rolls off very gradually at large values of frequency whereas the phase for 2I drops off rapidly as the frequency is increased above ω_{BW}. It is not surprising that this case (2I) received a poor pilot rating (PR=8) considering that attempts at aggressive tracking result in a closed loop divergence. Hence, we have evidence that the ability of the pilot to attain good closed loop regulation without threatening stability depends not only on

1) The value of bandwidth, ω_{BW}; but also on

2) The shape of the phase curve at frequencies above ω_{BW}.

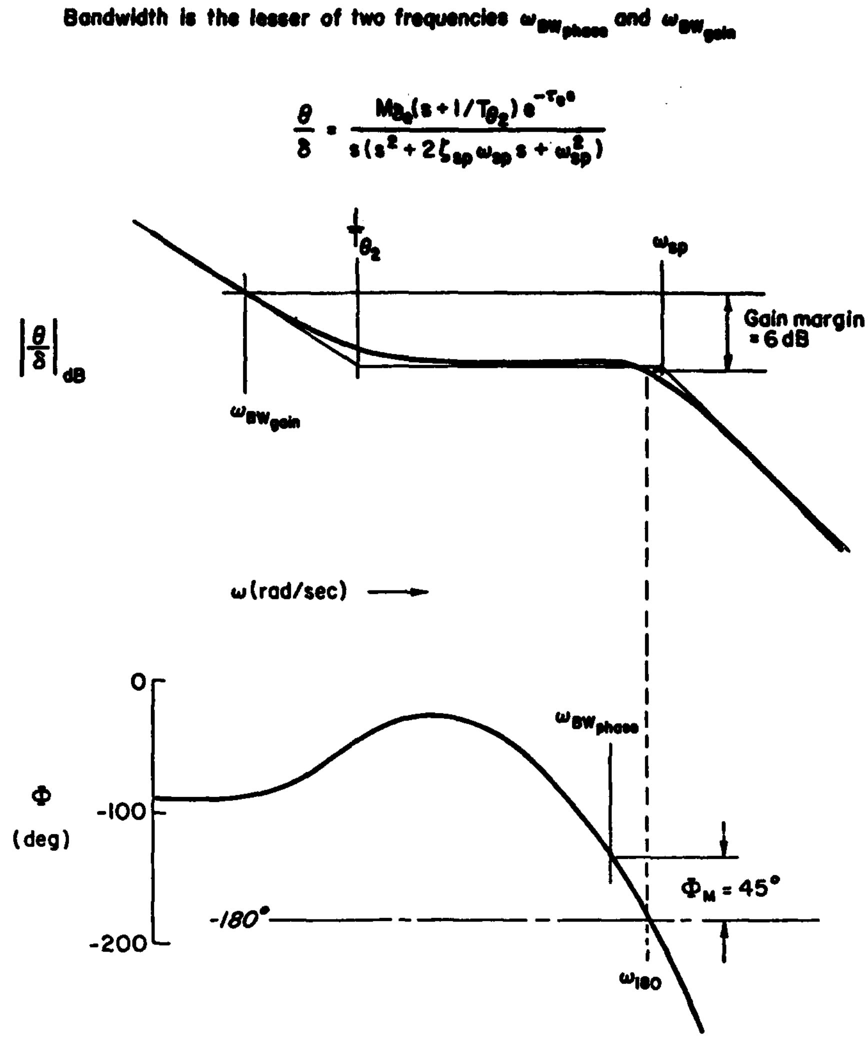


Figure 2. Definition of Bandwidth Frequency, ω_{BW}

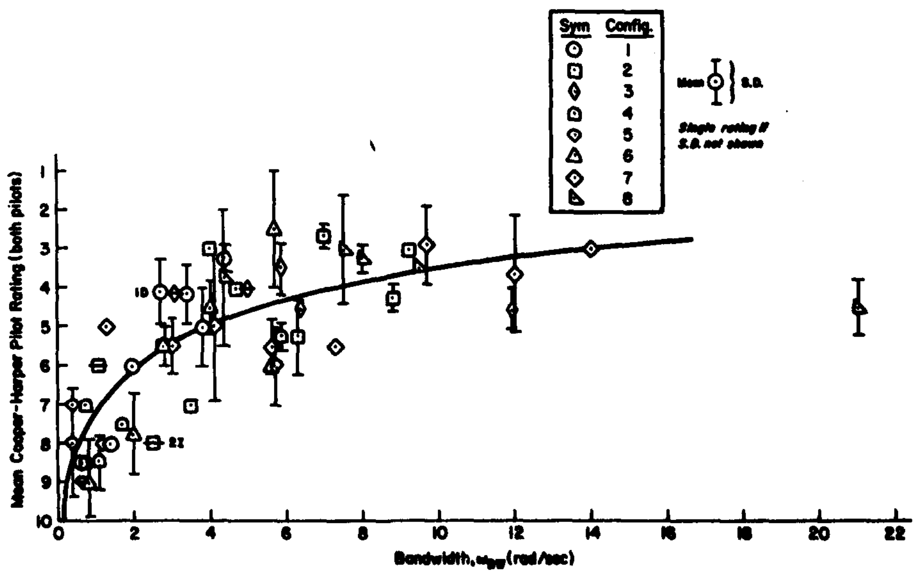


Figure 3. Comparison of Neal-Smith Data with Bandwidth (Mean Ratings)

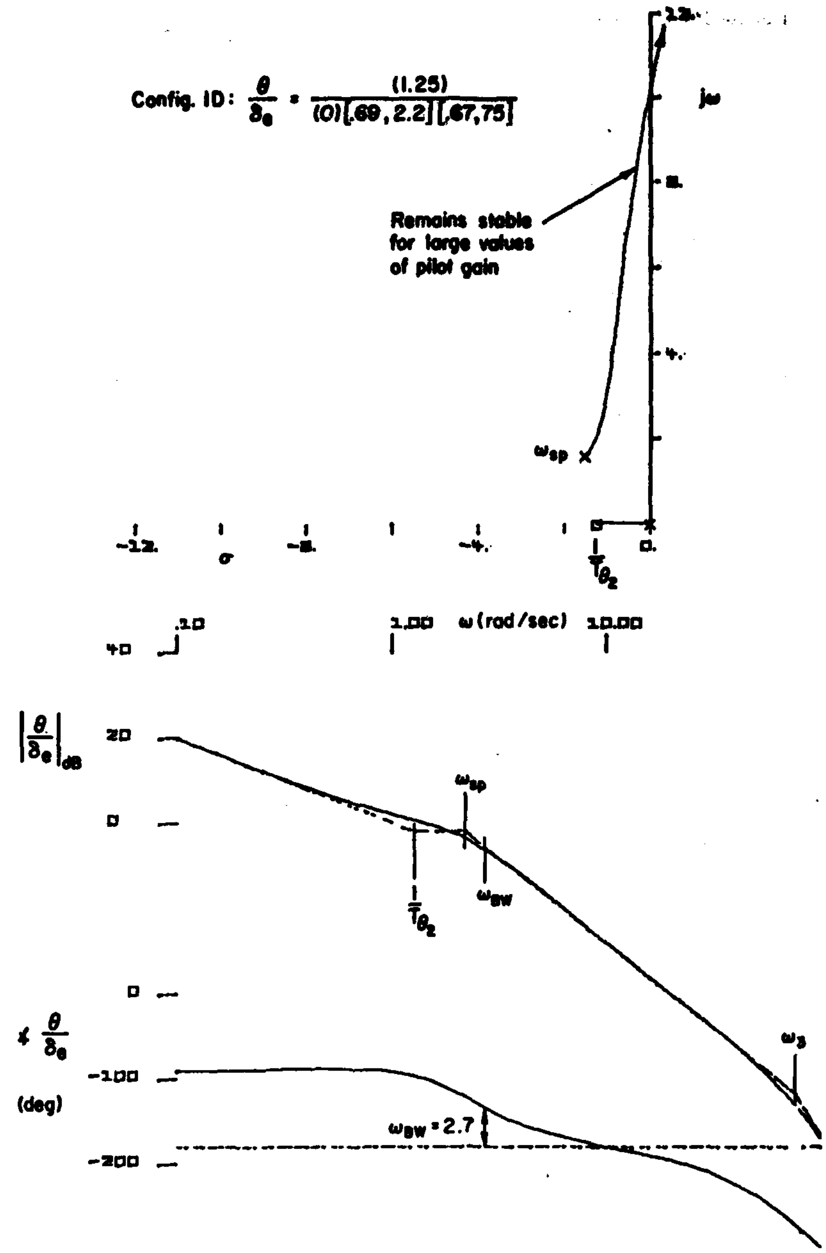


Figure 4a. Level 1/2 System of Neal-Smith (1D): ω_{BW} = 2.7 rad/sec, Mean PR = 4.1

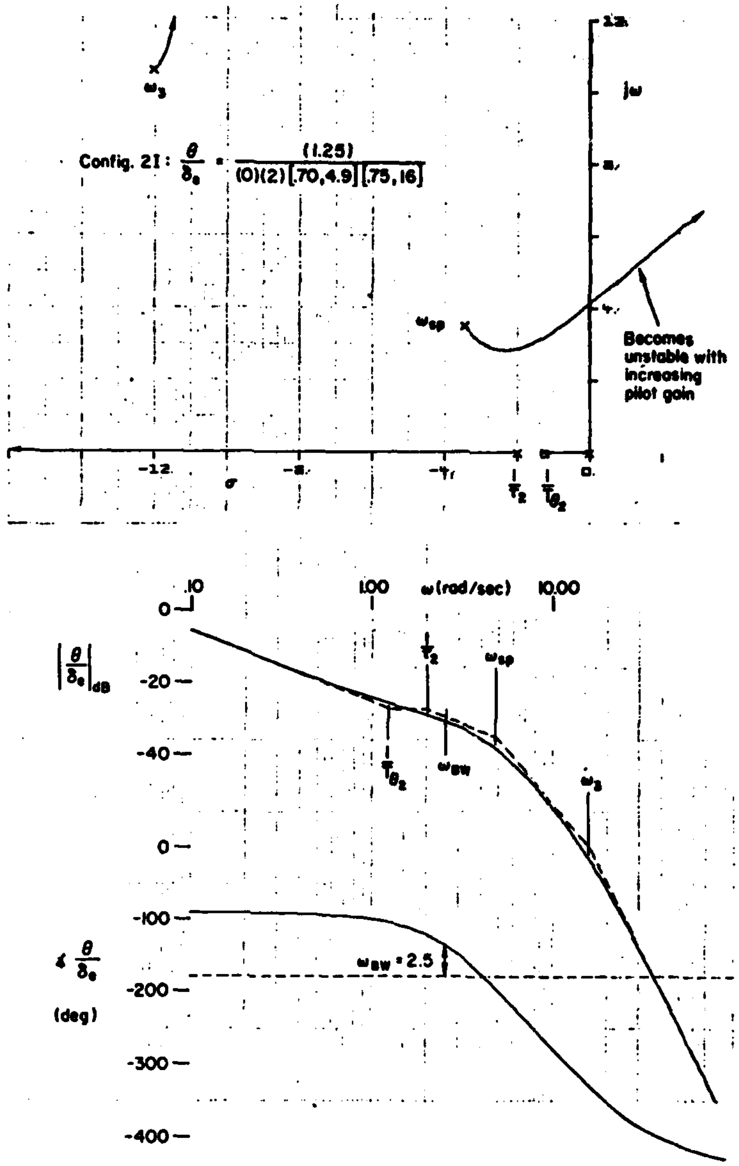


Figure 4b. Level 3 System of Neal-Smith (2I);
ω_{BW} = 2.5, Mean PR = 8.0

Rapid rolloffs in phase are well represented by a pure time delay $e^{-j\omega\tau}$. Accordingly, both of the key factors noted above will be accounted for by plotting pilot rating data on a grid of ω_{BW} vs. τ. This is done for the Ref. 1 data (which was plotted vs. ω_{BW} alone in Fig. 3) as shown in Fig. 5. The scatter is seen to be considerably reduced and the data are reasonably well separated into Level 1, 2, and 3 regions*. The values of τ used in this plot were obtained from lower order equivalent system fits of the higher order system transfer functions (Ref. 5). The lower order equivalent system form was:

$$\frac{\dot{\theta}}{\delta} = \frac{(s + 1/T_{\theta_2})e^{-\tau_e s}}{s^2 + 2\zeta_e\omega_e s + \omega_e^2} \tag{2}$$

This is an unnecessarily complex way to obtain a measure of the shape of the phase curve above ω_{BW}. A much simpler approach is to note that the change in phase due to a time delay is a linear function of frequency, i.e. $\Delta\phi = \tau\omega$. To the extent that the rolloff in phase beyond -180 deg can be attributed to τ_e in Eqn. 2, we can estimate τ_e in the vicinity of some frequency ω_1 as:

$$\tau_p = -\frac{\phi_1 + 180^\circ}{57.3\omega_1} \tag{3}$$

Where ω_1 is some frequency greater than the frequency for neutral stability and the symbol τ_p represents the estimate of τ_e. Correlations between τ_e and τ_p for the combined Ref. 1 and 6 data resulted in a correlation coefficient of 0.9$^\phi$6. Thus, there is very good evidence that τ_p can be used in place of τ_e in Fig. 5 as shown in Fig. 6. These results are reasonably encouraging with the exception of a number of Level 2 ratings at high values of bandwidth. The abbreviated pilot comments (taken from Refs. 1 and 7) indicate that abruptness and oversensitivity become a problem when ω_{BW} is large. This was especially true of the Ref. 7 pilot ratings (given in parenthesis in Fig. 6) A possible upper boundary on ω_{BW} is shown in Fig. 6 to account for this problem. This boundary is considered tentative because the issue of overresponsiveness is not completely understood at this time. A broader data base is felt to be necessary to verify the results concerning an upper limit on ω_{BW}.

In the air-to-air tracking experiment reported in Ref. 4, an upper limit on ω_{BW} could not be established for the wings-level turn maneuver. These data are shown in Fig. 7 where ω_{BW} is the bandwidth of the heading response in the wings-level turn mode. (The wings-level turn mode consisted of commanding yaw rate changes with the rudder pedals with zero bank angle. This was made possible by the use of a direct side force control). There does seem to be a trend towards acceptance of abruptness when tracking a target aircraft. For example, Configuration 13 in the Ref. 1 experiments was rated 7 and 5.5 due to "excessive sensitivity." However, in the followup experiment (Ref. 7) with a target aircraft, Configuration 13 was rated a 2 on two separate evaluations. At first glance this would seem to be an idiosyncrasy of different pilots and a different experiment. However, the target aircraft was removed during the Ref. 7 experiment and the rating went from 2 back up to 7 (see ◇ in Fig. 6).

The data correlations in Fig. 6 represent up and away flight and are appropriate for generating boundaries for Category A in MIL-F-8785C (Ref. 8). Data for Category C (approach and landing) may be found in Ref. 6. These data are correlated with ω_{BW} and τ_p in Fig. 8. The upper boundary on ω_{BW} for Level 1 is considered tentative for the reasons discussed above.

POSSIBLE SHORTCOMINGS

Definition of ω_{BW} for Shelf-Like Frequency Response

Responses which are gain-margin-limited tend to have shelf-like amplitude plots as shown in Fig. 9. With such systems a small increase in pilot gain results in a large change in crossover frequency and a corresponding rapid decrease in phase margin. The decrease in phase margin becomes critical for attitude control when τ_p is moderately large (of order 0.1 to 0.2). The two configurations shown in Fig. 9 are taken from the Ref. 6 experiment. Applying the previously discussed definition of bandwidth, we find that both Configuration 5-6 and 5-7 are gain margin limited. Both configurations suffer from the same deficiency, i.e., moderate values of τ_e combined with a shelf-like amplitude curve which results in a very rapid decrease in phase margin with small changes in pilot gain. However, the 6 dB limit selected to define $\omega_{BW_{gain}}$ does not "catch" Configuration 5-6. While this configuration is correctly predicted to be Level 2 (PR = 6) on the basis of τ_p (see Fig. 8) the value of ω_{BW} is in the Level 1 region. Had we picked a slightly higher value of gain margin to define ω_{BW}, the bandwidth for Configurations 5-6 and 5-7 would be approximately equal. However, because of the nature of shelflike frequency responses, there will always be a case which can "fool" the criterion. An experienced handling qualities engineer would immediately recognize the shelflike shape and moderate τ as a significant deficiency. However, the purpose of a criterion is to eliminate such judgement calls. Nonetheless, it is not expected that this idiosyncrasy will result in problems with correlating or predicting pilot rating data inasmuch as moderate (Level 2) values of τ_p are required to get a rapid phase rolloff in a frequency region where the amplitude curve is flat.

---

*Here Level 1 → PR < 3-1/2; Level 2 → 3-1/2 < PR < 6-1/2; Level 3 → PR > 6-1/2 where PR refers to the Cooper Harper pilot rating scale.

$^\phi\omega_1$ was taken as twice the neutral stability frequency, i.e., $\omega_1 = 2\omega_{180}$. Hence,

$$\tau_p = -(\phi_{2\omega_{180}} + 180^\circ)/(57.3 \times 2\omega_{180})$$

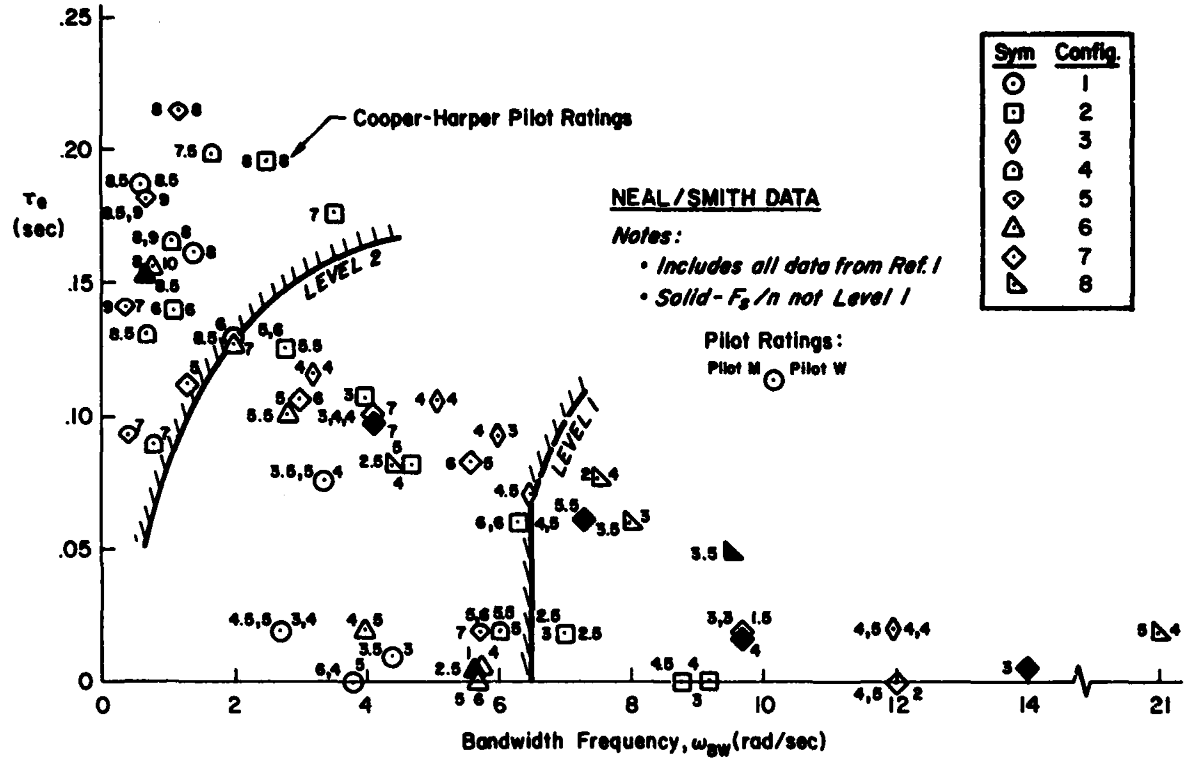


Figure 5. Correlation of Pilot Ratings with ω_{BW} and τ_e for Up and Away Flight (Ref. 1)

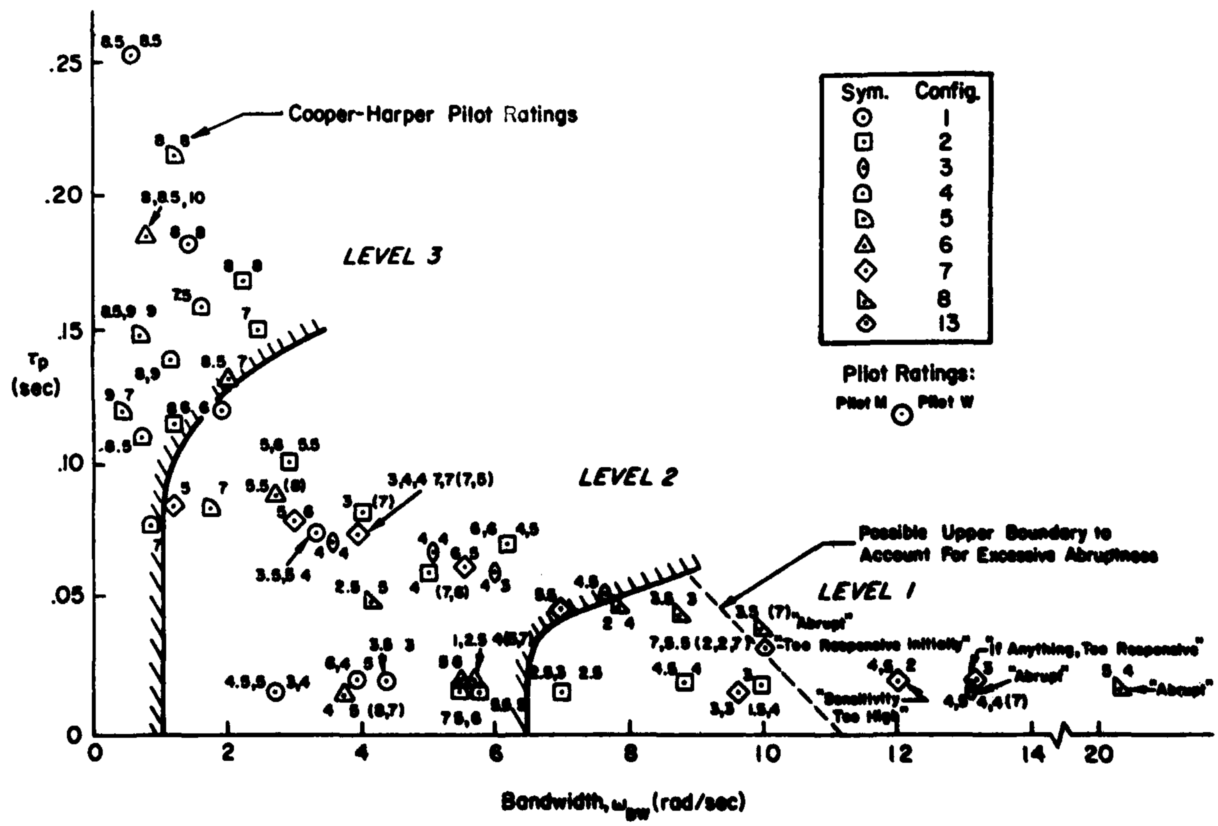


Figure 6. Correlation of Pilot Ratings with ω_{BW} and τ_p for Up and Away Flight (Data from Ref. 1, Ratings in Parentheses from Ref. 7)

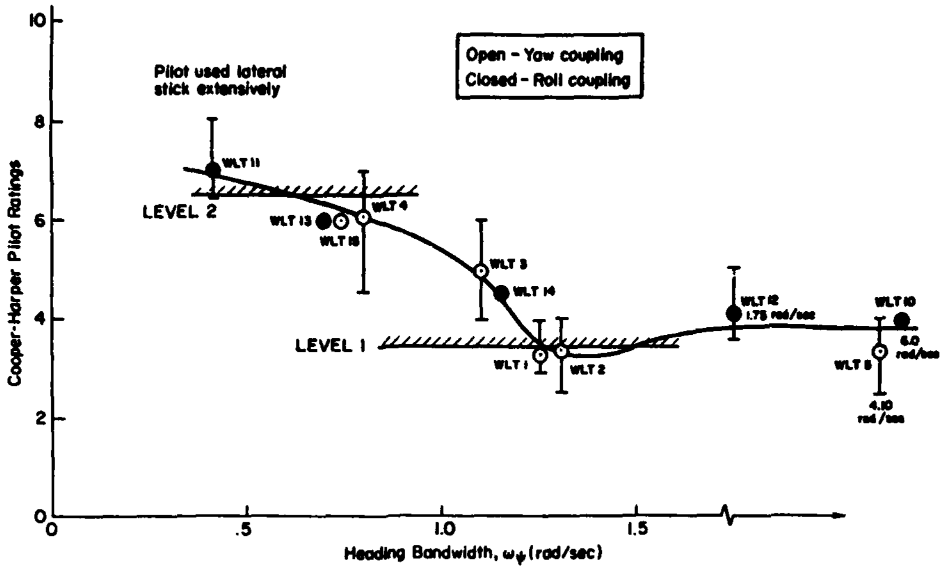


Figure 7. Correlation of Pilot Ratings with Heading Bandwidth; Wings-Level Turn Mode; Air-to-Air Tracking Task (from Ref. 4)

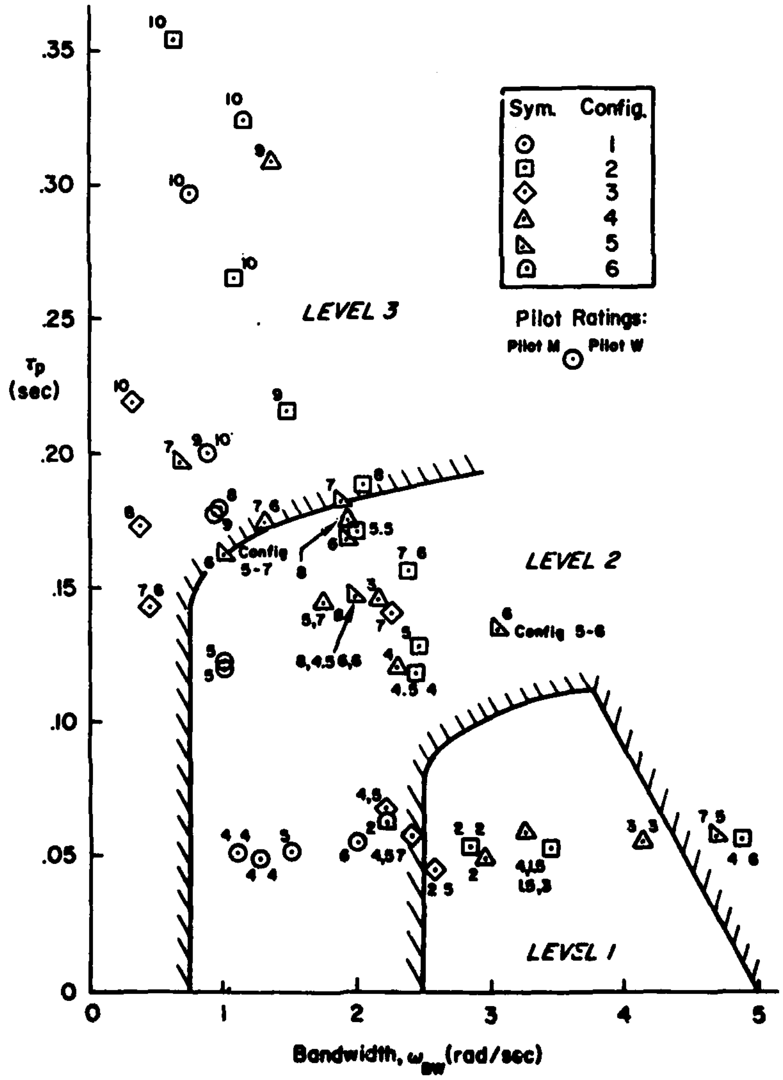


Figure 8. Correlation of Pilot Ratings with ω_{BW} and τ_p for Approach and Landing (Reference 6 Data)

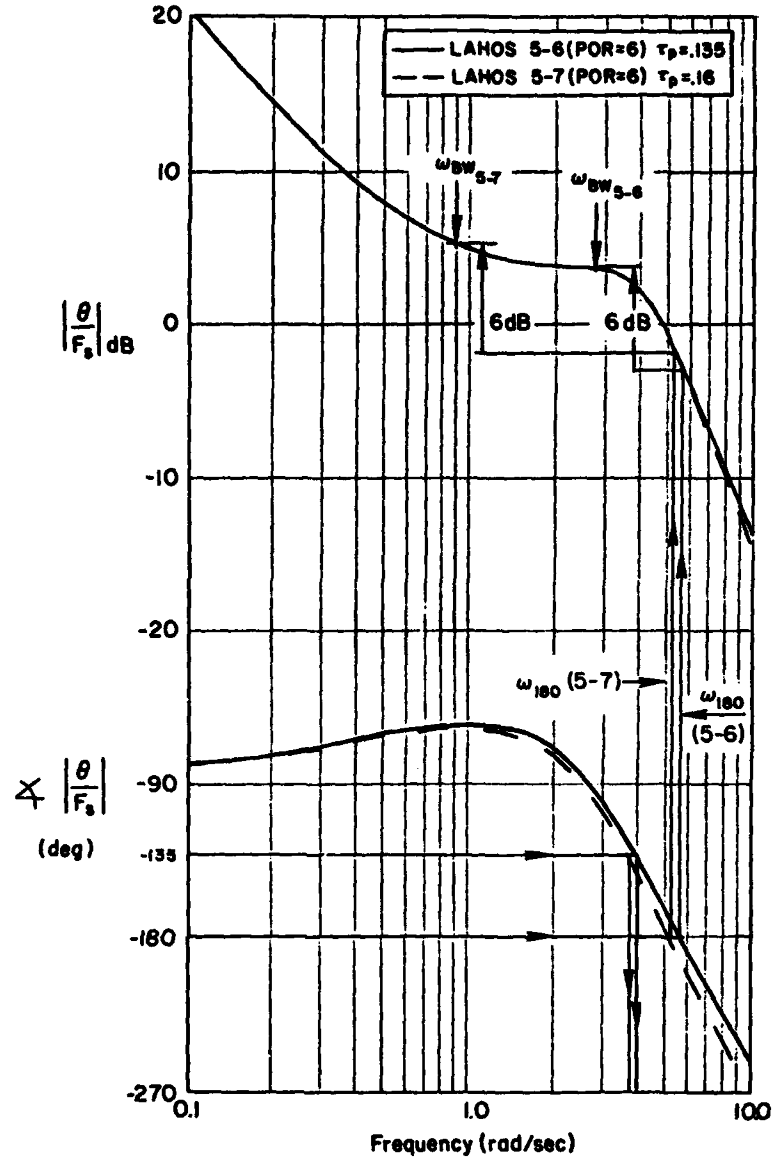


Figure 9. Large Difference in Bandwidth Due to Shelf in Amplitude Plot Combined with Moderate Values of τ_p

Measures of ω_{BW} and τ_p

ω_{BW} and τ_p are easily obtained when the frequency responses are available. However, the frequency responses themselves must be obtained from simulation or flight test data — e.g., as in the case of the Ref. 4 flight test of Direct Force Control modes. In that program, it was found that excellent frequency responses could be obtained by fast Fourier transforming flight test data. In particular, pilot generated frequency sweeps worked very well. A typical frequency sweep and the resulting Bode plot are shown in Figs. 10 and 11 respectively. The instrumentation required to obtain this data was minimal, consisting of a yaw rate gyro and pedal position transducer. Nonetheless, the data must be manipulated (via a Fast Fourier transform computer program) which is less desirable than reading parameters off a time response. However, the promise of a universally applicable parameter which works for highly augmented airplanes and is easily interpreted in terms of the pilot closed loop behavior seems an acceptable price to pay for a slight increase in complexity to define the parameters.

CONCLUSIONS

Bandwidth has been shown to be an effective parameter to discriminate between Level 1, 2, and 3 handling qualities for highly augmented airplanes. It was found that the shape of the phase curve above ω_{BW} is a key factor. Accordingly, the final criterion involves boundaries drawn on a grid of ω_{BW} vs. τ_p, where τ_p is an estimate of the pure time delay and is used here to define the phase curve shape for $\omega > \omega_{BW}$. Pilot rating data has been correlated for pitch attitude control and for a wings level turn direct force control mode with good results. However, there are some questions that need to be answered before it can be concluded that the criterion is universally applicable in its present form.

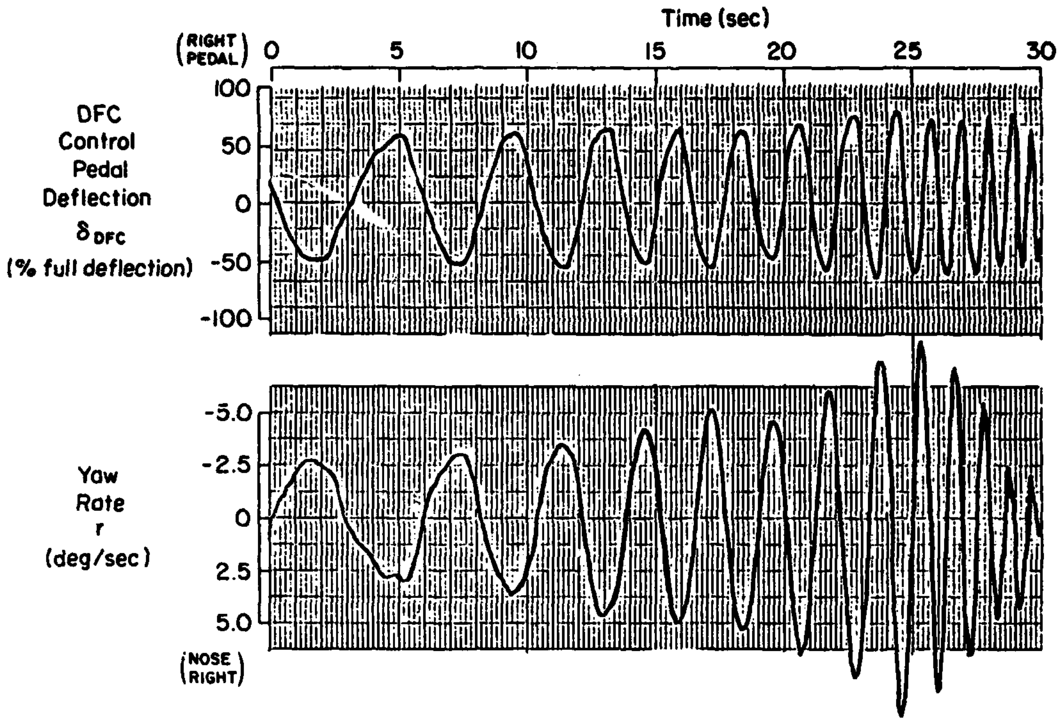


Figure 10. Typical DFC Control "Frequency Sweep" and Response for Configuration Identification (Configuration WLT 2, Ref. 4)

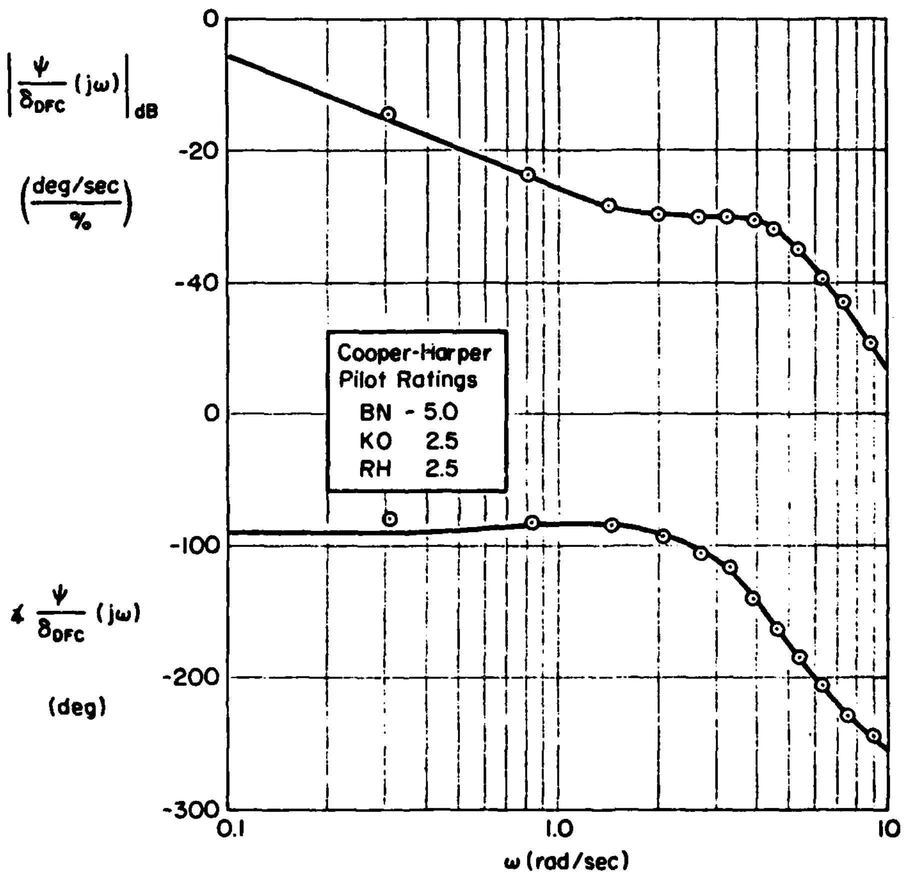


Figure 11. Fourier Transformed Heading Response Resulting From Frequency Sweep Shown in Fig. 10

REFERENCES

1. Neal, T. Peter, and Rogers E. Smith, An In-Flight Investigation to Develop Control System Design Criteria for Fighter Airplanes, AFFDL-TR-70-74, Dec. 1970.

2. Ashkenas, Irving L., Roger H. Hoh, and Samuel J. Craig, Recommended Revisions to Selected Portions of MIL-F-8785B(ASG) and Background Data, AFFDL-TR-73-76, Aug. 1973.

3. McRuer, Duane, Irving Ashkenas, and Dunstan Graham, Aircraft Dynamics and Automatic Control, Princeton University Press, Princeton, N.J., 1973.

4. Hoh, Roger H., Thomas T. Myers, Irving Ashkenas, Robert F. Ringland, and Samuel J. Craig, Development of Handling Quality Criteria for Aircraft with Independent Control of Six Degrees of Freedom, AFWAL-TR-81-3027, Apr. 1981.

5. Wood, J. R., and J. Hodgkinson, Definition of Acceptable Levels of Mismatch for Equivalent Systems of Augmented CTOL Aircraft, McDonnell Douglas Corp., MDC A6792, Dec. 1980.

6. Smith, Rogers E., Effects of Control System Dynamics on Fighter Approach and Landing Longitudinal Flying Qualities, AFFDL-TR-78-122, Vol. I, Mar. 1978.

7. Boothe, Edward M., Robert T. N. Chen, and Charles R. Chalk, A Two-Phase Investigation of Longitudinal Flying Qualities for Fighters, AFFDL-TR-74-9, Apr. 1974.

8. Military Specification, Flying Qualities of Piloted Airplanes, MIL-F-8785C, Nov. 1980.

ACKNOWLEDGMENTS

The work reported here was sponsored by the U.S. Air Force, Wright Aeronautical Laboratories, under Contracts F33615-78-C-3615 and F33615-80-C-3604.

HANDLING QUALITIES ASPECTS OF CTOL AIRCRAFT WITH ADVANCED FLIGHT CONTROLS

by

D. Hanke
K. Wilhelm
H.-H. Lange

Institut für Flugmechanik

Deutsche Forschungs- und Versuchsanstalt
für Luft- und Raumfahrt e.V. (DFVLR)
3300 Braunschweig Flughafen, West Germany

SUMMARY

This paper describes the problems which occur in applying the existing MIL-F-8785C Short-Period-Frequency Requirements to DLC-enhanced aircraft in flight path control situations. Recent test results indicate that the MIL-Spec. boundaries are only pitch related and not applicable to path control problems.

Further, based on DLC investigations carried out with the DFVLR-HFB-320 In-Flight-Simulator a new generalized flight path control criterion is proposed.

This criterion considers the multiloop landing approach situation characterized by the pitch inner loop and the altitude outer loop. The criterion philosophy is based on the frequency separation of the two control loops necessary for good handling characteristics. In-flight simulation indicates that this frequency separation is reduced by direct lift control, which leads to severe handling problems, especially for aircraft with poor pitch dynamics. The flight path to pitch attitude phase at the frequency of pilot-closed inner loop was selected as criterion parameter in representing the pitch/heave harmony or loop separation. This phase criterion is suitable for conventional or DLC-enhanced aircraft. A simple conversion to the MIL-$\omega^2_{nsp}/n/\alpha$-criterion for conventional aircraft is possible if n/α is interpreted as $-Z\alpha/g$.

In-flight investigations carried out by DFVLR using a rate command/attitude hold (RC/AH) system to augment pitch inner loop show that the pilot exhibits discrete control behaviour and open loop type control techniques. In general, RC/AH-systems lead to very low pilot activity and, combined with DLC, flight path control is improved. Due to this, new systems oriented handling qualities criteria have to be developed.

1. INTRODUCTION

The relatively recent implementation of complex control systems incorporating high authority and extensive command and stability augmentation into current aircraft leads to the requirement for new or revised handling qualities criteria development.

In a number countries significant research has been undertaken to solve these problems. The contents of this paper will be adressed to the very specific problem of precise flight path control of large transport aircraft on landing approach.

In particular, the handling qualities investigations related to direct lift control (DLC) application for flight path control enhancement carried out by DFVLR with the HFB 320 in-flight simulator [1-7] have resulted in better understanding of pilot/vehicle behaviour in path control situations and have led to a new flight path control criterion proposal. Covered in this paper are only DLC-systems acting as manoeuvre enhancement devices due to pitch inputs (pitch-path coupling). Systems decoupling pitch or path show completely different control behaviour compared with conventional aircraft and therefore lead to the requirement for special handling qualities criteria [2, 8].

2. FLIGHT PATH CONTROL PROBLEM

Large transport aircraft development shows a trend toward lower n/α-values due to high wing loading, which leads to sluggish flight path (altitude) response (s. fig. 1). Handling quality problems arise particularly during the landing approach, where precise flight path control is required. To overcome these problems the application of direct lift control appears to offer a promising solution. DLC has been discussed, investigated and partly realized during the last 15 years, so it is not new. But an appropriate handling qualities criterion which is able to describe the flight path control factors influencing pilot opinion independent of system design is still nonexistent.

So DFVLR investigations have been directed toward the development of a general flight path control criterion which can be used for both conventional and direct lift enhanced aircraft.

Today the only short period flight path control criterion specified in the MIL-F-8785 C [8] addresses the short-period frequency and vertical acceleration sensitivity requirements, which give a relationship between pitch and vertical acceleration aircraft response. In the past these requirements have been verified in many applications for conventional aircraft.

It is well known that the parameters as defined in the manoeuvre criterion cannot be used adequately for DLC-systems because vertical accelerations can be produced by DLC without angle of attack variations, so that the n/α-value approaches infinity.

Nevertheless, all the flight path control related 'knowledge' inherent in the manoeuvre criterion boundaries should be used for a new criterion by using a new more generalized parameter than n/α.

However, the data evaluation of DFVLR DLC-investigations and the attempt to correlate the results with the manoeuvre criterion boundaries led first of all to the unexpected result that the manoeuvre criterion boundaries are not representative of flight path control or vertical acceleration boundaries. What the manoeuvre boundaries really are and how the misinterpretation occured will be discussed in the next chapter.

3. A CRITICAL REVIEW OF THE MIL-F-8785C SHORT-PERIOD-FREQUENCY REQUIREMENTS

3.1 LIMITATIONS IN MIL-SPEC. PARAMETERS

The manoeuvre criterion of the MIL-F-8785 C is based on the CAP-Criterion established by Bihrle [10]. Bihrle used the ratio of initial pitch acceleration to steady state load factor $\ddot{\theta}_{(o)}/\Delta n_{zss}$ as a flying qualities criterion parameter (CAP) for aircraft manoeuvering, which correlates well with pilot ratings.

To simplify criterion application the above CAP was replaced by the 'equivalent' aircraft related parameter $\omega_{nsp}^2/n/\alpha$. This 'equivalence' was obtained by purely formal computation using short period approximation valid for conventional aircraft with negligible control system dynamics.

This approximation was implemented in the MIL-Specifications which, in retrospect, was overly restrictive because this parameter narrows down the application much more than the basic CAP formulation. Further, the Equivalent Systems Approach for highly augmented aircraft [11] is based upon this equivalent parameter to comply with the MIL-Specification, comparing equivalents of equivalent values. As a result, the general problem related to the use of $\omega_{nsp}^2/n/\alpha$ remains unresolved by this approach.

In general, the manoeuvre criterion parameter n/α, as shown in figure 2, has been interpreted as a parameter influencing mainly the short period flight path response. The lower n/α-boundary has been selected to characterize unacceptable flight path control. n/α is explicitly defined as acceleration sensitivity by MIL-F-8785 C. The prequisites for the validity of equivalent parameters are the following relationships between pitch dynamics, CAP and flight path response due to pitch input.

$$\dot{\theta}/_{\delta e} = \frac{M\delta e \ (s + {}^1/T_{\theta 2})}{s^2 + 2\xi\omega_{nsp}\ s + \omega_{nsp}^2} \tag{1}$$

$$CAP = \frac{\ddot{\theta}(o)/\delta}{n_{zss}/\delta} = \frac{\omega_{nsp}^2}{\frac{V_o}{g} \cdot {}^1/T_{\theta 2}} = \frac{\omega_{nsp}^2}{n/\alpha} \tag{2}$$

$$\gamma/_{\theta} = \frac{1}{T_{\theta 2}\ s + 1} \tag{3}$$

$$T_{\theta 2} = -\frac{1}{Z_w} \tag{4}$$

The numerator time constant $T_{\theta 2}$ in the short-term pitch rate response (Eq. (1)) physically represents the pitch rate overshoot behaviour.

In addition, $T_{\theta 2}$ physically represents vertical acceleration sensitivity n/α (Eq. (2)) and the short-term flight path lag due to pitch inputs (Eq. (3)).

Now, for advanced control systems the pitch rate overshoot behaviour can be changed independently of flight path lag. Further, through DLC application flight path lag will be changed without changing the pitch rate overshoot. Therefore, the important relationship $(V_o/g)\ (^1/T_{\theta 2}) = n/\alpha$ on which the manoeuvre criterion is based, is invalid.

Because in the MIL-Spec. n/α has the physical meaning of vertical acceleration sensitivity, the independent influence of pitch rate overshoot (numerator time constant $T_{\theta 2}$) cannot be accounted for.

This problem does not exist using the original CAP definition due to the fact that the influence of the numerator time constant is included in initial pitch acceleration response. Using DLC, independent n/α-variations are possible with either n/α going to infinity (this would not be a problem using the original CAP definition n_{zss}/δ) or steady state n/α remaining unchanged. However, high frequency vertical acceleration response (s. figure 3) might lead to unacceptable response characteristics, an effect which cannot be covered by the MIL-Spec. requirements.

3.2 EQUIVALENT SYSTEMS APPROACH

With the Equivalent Systems Approach for longitudinal control the essential characteristics of the higher order system are represented by a lower second order system [11]. Using a Bode plot matching technique, the two equivalent parameters equivalent frequency ω_{ne} and equivalent steady state load factor due to angle of attack n/α_e are computed. In addition, an equivalent time delay is used to match the phase lag of higher order systems.

Equivalent parameters can be used on existing manoeuvring criterion. But matching problems occur by using basic aircraft n/α with $^1/T_{\theta 2}$ fixed or free. With $^1/T_{\theta 2}$ free, and computing n/α_e as equal to $V_o/g\ (^1/T_{\theta 2e})$ good correlations are obtained but with physically meaningless n/α_e-values. By A'Harrah [12] it was concluded that not n/α but $^1/T_{\theta 2}$ is the appropriate parameter, which actually means that n/α is an equivalence for pitch rate overshoot. But this is in contrast to the MIL-F-8785 C definition of n/α. To overcome the problem of getting physically meaningless n/α-values, A'Harrah [13] proposed making simultaneous matches of normal load factor and pitch rate characteristics to provide values of equivalent n/α which are more in line with wind-tunnel derived values than are the pitch-rate only match values.

However, this again generates the old confusion of whether $^1/T_{\theta 2}$ or n/α is the appropriate parameter. For future applications of the manoeuvre criterion this question must be definitively answered. Due to our own research there is substantial experimental support for selecting the numerator time constant as the correct parameter. This will be verified in the next chapter.

3.3 PHYSICAL INTERPRETATION OF MIL.-SPEC. PARAMETERS

At first the ω_{nsp} and n/α-data of thirty existing aircraft on landing approach were computed and compared with the manoeuvre criterion. This result is illustrated in figure 4 showing that the n/α-value variations comparing similar sized aircraft are minor.

The differences are caused mainly by different pitch inertias between small to large aircraft. The same results can be obtained by using the simple approximation

$$n/\alpha = -\frac{V_o}{g} \cdot Z_w \qquad (5)$$

$$n/\alpha = \frac{V_o}{g} \cdot \left(\frac{\rho}{2} V_o \frac{S}{m} \cdot (C_{L\alpha} - C_{D_o})\right) \qquad (6)$$

for horizontal flight conditions.

Assuming weight equal to lift,

$$W = mg = L, \qquad (7)$$

it follows that

$$n/\alpha = \frac{(C_{L\alpha} - C_{D_o})}{C_{L_o}} \qquad (8)$$

This shows that in landing approach, physically possible values vary only between 3 and 6 for CTOL transport aircraft. n/α-values below 3 are impossible for aircraft with aerodynamic lift. Further, the figure show that larger aircraft tend to have lower pitch dynamics than low n/α-values.

It seems to be very unlikely that n/α could have a big influence on handling qualities compared with pitch dynamics (short period frequency) because, due to physical reasons, the n/α-values are nearly identical for similar sized aircraft.

A comparison of the pure pitch criterion from Neal-Smith [14] and the manoeuvre criterion over a wide range of aircraft parameters is shown in figure 5. This figure shows that the Neal-Smith boundaries correlate very well with the manoeuvre criterion boundaries.

The Neal-Smith lead/lag-zero line is within the MIL-boundaries, the upper boundary correlate with oversensitive (phase lag > 20 deg) aircraft behaviour and the lower boundary with too sluggish response (phase lead > 40 deg). An important difference is seen for the low n/α-boundary where the Neal-Smith criterion set the resonance boundary influenced by numerator time constant. It is interesting that parameter combinations shown in the right plane of the figure are acceptable using the Neal-Smith criterion. However MIL-damping requirements would reject these configurations. But this region is more theoretical than practical.

The comparison shows that, in reality, the handling qualities parameters used in the MIL requirements are related to pitch axis. The physical meaning of n/α is not vertical acceleration sensivity but numerator time constant influencing pitch rate overshoot and initial pitch acceleration too.

Further, only with this interpretation of n/α the equivalent system approach is able to comply with MIL-boundaries. A typical result which support this conclusion is shown in figure 6 obtained by Mooij [15]. Investigated was a rate command/ attitude hold system where the numerator time constant of the pitch rate command model in forward loop was varied with <u>constant n/α</u> of the basic aircraft. The results show good correlation with the low n/α boundary of MIL requirements if equivalent n/α_e is used which represents numerator time constant. Further, variation of command model frequency (numerator time constant fixed) results in correspondence with the MIL-boundary.

3.4 CONCLUSIONS IN USING THE MIL.-SPEC. REQUIREMENTS FOR FLIGHT PATH CONTROL PROBLEMS

Summarizing the results of the discussion the following conclusions can be drawn:

1. The decision to use short period approximation related parameters $\omega^2_{nsp}/n/\alpha$ instead of $\ddot{\theta}(o)/\Delta n_{zss}$ in the MIL.-Spec. short-period frequency requirements leads firstly to unnecessary limitations in application and secondly to misinterpretation of the physical meaning of the n/α parameter.
2. There are strong indications that the manoeuvre criterion represent only pitch dynamic parameters, short period frequency and pitch rate numerator time constant and not vertical acceleration sensivity.
3. To avoid confusion of the physical meaning of criterion parameters, n/α or n/αe should not be used as equivalent parameters for pitch rate numerator time constant.
4. Generally, the manoeuvre criterion and the Neal-Smith criterion cannot cover acceleration response characteristics on short term aircraft response.
5. The proposal of using the Equivalent System Approach to compute an equivalent n/α-value by simultaneous matching of pitch rate and vertical acceleration response indicates that both responses are important for handling qualities. But in this approach n/α_e is not clearly defined and the old controversy n/α or $^1/T_{\theta 2}$ is revived.
6. An additional acceleration response criterion independent of pitch response is necessary.
7. It should be noted that pilot ratings on which the existing pitch criteria boundaries are based include path or vertical acceleration influences. Even though a basic pitch/ acceleration coupling exists, it has not been possible, in the past, to alter the acceleration response because it was fixed by the aircraft's wing design and flight conditions. So the pitch dynamics were adapted to the given vertical acceleration response.

   With the implementation of DLC, however, it is possible to change the acceleration response independent of pitch behaviour. The influence of acceleration response can thus be investigates in a much more direct manner than was possible in the past.

4. DEVELOPMENT OF A NEW FLIGHT PATH CONTROL CRITERION

4.1 TEST DESCRIPTION

As mentioned in chapter 2, the flight path control criterion have been developed mainly for direct lift control application. The results are based on investigations published in [6] where the flight characteristics of a large transport aircraft with various direct lift

control configurations were simulated in-flight using the DFVLR-HFB 320 variable stability aircraft (fig. 7). The direct lift control concepts investigated consist of simultaneous deployment of elevator and wing spoilers when the pilot inputs a command in pitch axis. The concept therefore gives spoiler deflection as a direct function of pilot control input. The principle is shown in fig. 8.

The pilot's pitch input is routed to the spoiler system via a wash-out term. The elevator and spoiler actuator dynamics are represented by a first order system. The pitch and normal acceleration behaviour are critically dependent on the choice of wash-out values and spoiler actuator dynamics [5]. The values were chosen following a pilot-in-the-loop analysis using the Neal and Smith criterion [14] applied to flight path control. In particular, the spoiler time constant was set to values avoiding too rapid acceleration response.

A comparison was made between an aircraft without direct lift control and four direct lift control configurations differing only in the DLC effectiveness for a fixed value of wash-out time constant.

The elevator control gain was chosen to give constant load factor ('g') per unit control displacement, or control force, to avoid changing the influence of this aircraft handling parameter even further.

The evaluation task was to perform ILS approaches using instruments and "raw" ILS data and to initiate a go around at 500 ft above ground level.

A test-flight mission consist of 4 - 5 approaches including the whole approach pattern with simulated aircraft dynamics. A total of 56 approaches with two pilots were flown.

4.2 TEST RESULTS

The pilot-aircraft performance was assessed against the requirements set for the ILS landing approach task. The following ILS-approach performance criteria were to be satisfied: glideslope hold accuracy of $\pm$ 0.5 Dot (total range $\pm$ 2 Dot). localizer hold accuracy of $\pm$ 0.5 Dot (total range $\pm$ 2 Dot) and speed control accuracy of $\pm$ 5 kts.

The mean values and standard deviations of the glideslope error, localizer error and speed deviation indicate that for all the investigated configurations both pilots satisfied the performance and did not reach or exceed the allowable limits in any case. This means that no configuration in particular stands out on the basis of overall performance.

The assessment of the configurations investigated was by an effort scale in which the pilot rated his effort from 0 (no effort) to 10 (high effort), the Cooper-Harper scale was also used, which relates the performance of the pilot-aircraft system and the work-load. The summarized result shown in figure 9 was that with increasing DLC effectiveness both pilots rated the system worse than the basic aircraft without DLC. Especially the configurations with high DLC effectiveness were rated unacceptable. This result was surprising and unexpected. Therefore the pilot-aircraft system was analized in more detail.

4.3 PILOT-AIRCRAFT SYSTEM ANALYSIS

The pilot technique applied for CTOL aircraft during an ILS approach is that he maintains the earth fixed flight path primarily by pitch attitude changes. In doing so he completes two control loops, an inner stabilizing pitch and an outer maintaining altitude. Altitude error is converted by the pilot into required pitch changes i. e. pitch commands, which are then realized in the inner control loop. Fig. 10 shows these relationships in control loop format, a structure which has been confirmed by many investigations and can be found in many references (see in particular [16, 17]).

From the above mentioned relationships the pilot has two tasks in the short period range:

1) Primarily, pitch attitude hold
2) Flight path hold i. e. altitude hold via pitch attitude.

The handling qualities are thus defined by two dynamic characteristics of the aircraft:

1) Pitch dynamics θ/δ,
2) Altitude change dynamics due to pitch attitude changes, h/θ.

The achievable bandwidth of the h-loop of the closed system is dependent on the inner loop characteristics [17].

This relationship shows clearly the importance of the pitch dynamics for flight path control. The bandwidth separation of pitch- and h-loop is determined by the h/θ or γ/θ-transfer function, since h results from integration of γ with the approach speed V_o as constant factor.

For conventional aircraft with elevator control the transfer function γ/θ (short period approximation), under the assumptions

$$|Z_{\delta e} M_w| << |Z_w M_{\delta e}| \quad \text{and} \quad |Z_{\delta e}| << |M_{\delta e}| \quad \text{is:}$$

is

$$\gamma/\theta = \frac{1}{T_{\theta 2}\, s + 1} \tag{9}$$

with

$$T_{\theta 2} = -\frac{1}{Z_w}\,. \tag{10}$$

This means that the transfer function γ/θ is approximately of first order with time constant $T_{\theta 2}$ which is inversely proportional to the derivative Z_w, where

$$Z_w = -\left(\frac{\rho}{2} V_o \cdot \frac{S}{m} (C_{L\alpha} - C_{D_o})\right) \tag{11}$$

is an aircraft inherent quantity, which is fixed due to flight conditions.

From the literature it is known that the time constant affects the handling qualities of flight path tracking tasks [17]. In particular low Z_w-values (therefore large lag γ following θ changes) leads to handling problems.

The purpose of direct lift control is to reduce the lag to increase the flight path tracking bandwidth. It is not yet clear how small or large the lag can be as a function of the pitch dynamics without impairing the handling. When using direct lift control the above first order approximation for the γ/θ transfer function is no longer valid. The relationship is of a higher order.

Because the use of open loop DLC naturally also affects the pitch dynamics, the tested configurations were compared with existing pitch criteria to identify the cause of degradations in handling with increasing DLC effectiveness.

With increasing DLC the pitch response of the aircraft becomes more sluggish, and the magnitude of the pitch rate overshoot reduces.

A comparison of the responses with the pitch rate criterion for the landing approach [18], shows that all the configurations satisfy the criterion. Only configuration B impinges the lower criterion boundary (fig. 11).

Fig. 12 shows the results of applying the Neal and Smith criterion for a pitch control loop closed by the pilot. All the configurations satisfy the criterion i. e. all values lie within the PR = 3.5 boundary (satisfactory handling). However configuration D lies on the boundary of the criterion for maximum pilot phase lead.

Checks of other suggested criteria ([19] and [20] show that the degradation in handling is not due to the change in pitch dynamics, since all configurations satisfy the known criteria with PR = 3.5 or only just fail them.

4.4 *FLIGHT PATH CONTROL CRITERION*

It was shown in the previous section that small changes in pitch dynamics of the aircraft, due to the DLC system, were not the explanation for different handling ratings found during the tests.

From this it follows that the change in γ/θ dynamics was primarily responsible for the change in the handling ratings.

Fig. 13 shows the Bode diagram in amplitude and phase for the γ/θ transfer functions of all the tested configurations. The figure shows clearly that with DLC the phase of the γ/θ transfer function at higher frequencies reduces less steeply, or increases earlier. The amplitude relationship shows similar trends to the phase relationship.

The greater the phase reduction, the slower the flight path changes due to pitch attitude changes in a given frequency range, so that the phase of the γ/θ transfer function directly represents the coupling between pitch attitude and flight path due to control inputs.

In particular it is apparent that the phase relationship in the pitch control frequency range (inner control loop, short period pitch oscillation frequency) is changed drastically by DLC, and can even become positive (configuration B), which means that the flight path leads pitch attitude.

The bandwidth of the altitude control loop are easily found from the Nichols diagram (fig. 14) which shows the amplitude and phase with h/h_c-loop open and the inner pitch control loop closed. For inner loop closure the Neal-Smith method was used with pilot model as shown in figure 10. The closed loop bandwith was set to $\omega_{BW\theta}$ = 1.2 rad/s for landing approach. The bandwidth of the h-loop is defined by -90 deg phase and 0 dB amplitude of the closed loop system. In this it is assumed that the pilot desires unity gain (0 dB) and operates as a pure amplifier (Gain K_h) in the altitude loop.

From figure 14 h-loop bandwith was extracted giving the values listed below

Conf.	h-loop bandwith ω_{BWh} (inner loop closed)	Outer-inner loop bandwith ratio $\omega_{BWh}/\omega_{BW\theta}$ ($\omega_{BW\theta}$ = 1.2 rad/s)
A	0.32 rad/s	0.27
C	0.38 rad/s	0.32
D	0.50 rad/s	0.42
B	0.70 rad/s	0.58

It is apparent that DLC reduces the bandwidth separation (or bandwith ratio $\omega_{BWh}/\omega_{BW\theta}$ between the altitude control loop and the pitch control loop, which inevitably results from the phase relationships of the γ/θ transfer function. It seems that when the pilot applies the technique of controlling the flight path by pitch attitude, a clear separation of the bandwidths between pitch and altitude loops is preferred. This means that inputs to stabilize the pitch attitude (in the frequency range of the short period pitch oscillation) should not simultaneously lead to changes in the flight path.

First of all, good pitch characteristics are mandatory for flight path control. In addition, the bandwith separation of the inner and outer loops has to be taken into account.

Both too large or too small loop bandwith separation will lead to handling problems. As an appropriate parameter to describe the bandwith separation or the degree of coupling between pitch and flight path, the phase difference of the γ/θ-transfer function in the region of inner loop frequency ω_θ will be used.

This parameter will be defined as

$$\gamma/\theta\text{-Phase for } \omega = \omega_{nsp} \text{ or } \omega_\theta$$

written as

$$\boxed{\varphi(\gamma/\theta)\big|_{\omega} = \omega_{nsp}\ (\omega_\theta)} \qquad (12)$$

This, then, is the proposed criterion for flight path control.

This criterion is able to fulfill some fundamental requirements.

- it takes into account the pilot-aircraft inner and outer loop structure
- it is applicable for conventional and DLC enhanced aircraft
- it is not restricted to low order systems
- it is physically understandable
- it clearly separates pitch and heave motion
- it is easy to compute
- it is easy extractable from flight test data
- there is no need to compute equivalent n/α values

A further requirement is that the criterion be compatible with existing MIL-F-8785 C specifications. If the n/α of MIL requirements can be interpreted as $\Xi\alpha/g$ (although this is questionable) a very simple translation to the phase criterion is possible. Because the MIL-Spec. boundaries are based on conventional aircraft where the short period approximation is valid (i. e., $\gamma/\theta = 1/(T_{\theta 2}\ s + 1)$ the γ/θ-Phase for $\omega = \omega_{nsp}$ can simply be computed

by

$$\varphi(\gamma/\theta)|_{\omega = \omega_{nsp}} = -\arctan(T_{\theta 2} \cdot \omega_{nsp}) \qquad (13)$$

or using MIL-Spec. parameters

$$\varphi(\gamma/\theta)|_{\omega = \omega_{nsp}} = -\arctan\left(\frac{V_o}{g} \frac{\omega_{nsp}}{n/\alpha}\right). \qquad (14)$$

Figure 15 illustrates this translation of MIL-Spec. boundaries into γ/θ-Phase diagram. V_o/g was taken to be constant, since for all transport aircraft the approach speed is approximately the same. Typical approach speeds are between 130 - 140 kts. $V_o/g = 7$ s was chosen as a representative quantity.

The right plane boundaries of figure 15 comply with other results from [18] and with the investigated DLC configurations. But, as shown in chapter 3, at least the left hand boundary (F - G) has to be interpreted as numerator time constant-boundary of pitch rate transfer function and not as flight path boundary.

Although the transformation of MIL-boundaries into γ/θ-phase boundaries looks very promising, the MIL-Spec. boundaries are not usable because of different physical meaning of n/α (see chapter 3).

Because both criteria, pitch and flight path, must be fulfilled, another approach would be to combine the flight path phase criterion with the Neal-Smith pitch criterion. It has been found that large transport aircraft show no tendency to closed loop resonance, so a presentation as shown in figure 16 could be used to combine pitch and flight path control behaviour.

The area filled by existing aircraft is shown by the shadowed area, in which some typical aircraft are pointed out. It can be seen that so extremely different aircraft as the lifting body HL-10 and the large transport aircraft C-5A stay within -70 degrees lag to 40 degrees lead phase in pitch using the Neal-Smith criterion.

But the γ/θ phase remains within -80 degrees to -55 degrees path lag. This is due to the fact that the approach condition $C_{L\alpha}/C_{Lo}$ is not so different for all aircraft. Further, from this it can be concluded that, in the past, path control was not really a problem compared with pitch dynamics. Neither the γ/θ-low phase nor high phase lag boundaries are very well established because very few data due to DLC application are available.

It is recommended from the flight tests that for large aircraft with sluggish pitch response the flight path lag should not be lower than -30 degrees. Further it can be concluded that flight path lag of existing large transport aircraft with -50 degrees is adequate. Path control problems of large aircraft are not due to severe pitch path coupling but rather to absolutely unsatisfactory pitch response. This means that DLC will never improve path control if pitch control is inadequate.

Conclusions

The conclusions listed below are valid only for instrument approaches without Flight-Director and assume that the flight path changes result from pitch attitude changes, which is the normal case for CTOL aircraft.

1. The altitude hold of an aircraft is directly affected by the pitch hold characteristics.

2. A one-sided increase in bandwidth in the altitude loop through DLC leads to a degradation in handling. It follows from this that flight path control with sluggish pitch dynamics cannot be improved by the use of DLC.

3. To use the CTOL techniques (flight path control through pitch changes), a given bandwidth separation between pitch control and altitude control is required for good handling qualities.

4. An improvement in flight path control on aircraft with sluggish pitch dynamics can be achieved by DLC only if the pitch dynamics are also improved.

5. The phase difference between flight path and pitch attitude (γ/θ-phase) in the region of the frequency of the short period pitch oscillation can be used as a criterion for the bandwidth separation between the pitch attitude control loop and the altitude control loop, that is the coupling between pitch motion and vertical motion of the aircraft.

6. The γ/θ phase criterion can be applied to conventional aircraft as well as those with DLC. It can be translated directly into the manoeuvre criterion of MIL-F-8785 C if n/α is interpreted as $-Z_\alpha/g$.

5. FLIGHT PATH CONTROL WITH RATE COMMAND/ATTITUDE HOLD (RC/AH) SYSTEMS

The results shown in the previous chapters are related to more or less conventional systems, in other words aircraft requiring only conventional response and piloting technique to fly an ILS-landing approach.

This conventional technique is based on two-loop structure with clearly separated bandwith of each loop. In general, it has been shown that path response of large aircraft compared with pitch response is not bad. DLC application cannot improve path control because pitch/altitude coupling will be unfavourably changed. From this it is concluded that DLC must improve flight path control capability without degrading handling qualities by using systems with inner loop augmentation such as a rate command/attitude hold system.

In that case the inner pitch loop is overtaken by the control system so that there is no need for the pilot to stabilise pitch attitude or to counteract external disturbances. The manual flight path control loop structure will be simplified to a single altitude loop, flown by pitch attitude.

Flight tests with a RC/AH-system combined with DLC have been carried out using the DFVLR-HFB-320 in-flight simulator. The RC/AH-system function, as realized in the digital onboard computer, is shown in figure 17. For pilot commands a two axis sidegrip controller for pitch and roll was used (figure18).

The flight test results show that advanced systems like RC/AH-system for large transport aircraft require new systems oriented criteria because many assumptions on which the old criteria are based are no longer valid.

Firstly, as shown in figure 19, (which represents a typical landing approach with RC/AH-system under gusty conditions) the pilot control behaviour changes completely compared with conventional aircraft. The pilot inputs are 'pulse' type. Because the system holds the attitude there is no need for the pilot to be permanently active in the loop. The pilot tends to control in an open loop fashion; the loop is only closed after a certain pilot-determined threshold is exceeded. The aircraft response is simple, so the pilot can anticipate its behavior. The loop is only 'closed' by monitoring the aircraft response.

The RC/AH-system was combined with direct lift device as shown in figure 18 to investigate the pitch/path coupling without the inner loop situation. The pitch/path coupling was varied in directions; in one direction using negative DLC to enlarge the path lag due to pitch attitude commands, and in the other direction using positive DLC to reduce the path lag.

The results show that the pilots were not sensitive to negative lift effects, to the extent investigated, compared with the basic aircraft. But using the same amount of positive lift the ratings improved as expected. Figure 20 shows the culmulative percentage of time that the side-grip controller was used during approach versus the percentage of grip deflection in pitch and roll axes with and without DLC. The figure illustrates that about 50 - 60 % of approach time the grip is not used by the pilots for the RC/AH-system without DLC. With DLC the grip is not used about 75 % of time in pitch and roll. In addition, percentage of grip deflection is reduced.

It is interesting that the improvement in flight path control leads to lower pilot activity in the roll axis. Pilots commented that with DLC vertical speed variation could be initiated and stopped very precisely so there was more time to control the roll attitude.

The results gained from the RC/AH flight tests program can be concluded

1. DLC improves flight path control if the pitch inner loop is augmented by a RC/AH-system because the frequency separation of inner and outer loop has no relevance.

2. RC-AH-system characteristics lead also to lower activity in roll control.

3. RC/AH leads to generally low pilot activity, and the addition of DLC lessens his activity even further.

4. Improvements in longitudinal control lead to lower activity in roll control.

5. The pilot closes the loop after the system exceeds a given threshold set by the pilot. Due to predictable aircraft response, the pilot tend to act as open loop controller.

6. Nevertheless the γ/θ relationship remain important for path control. Boundaries for path control with augmented inner loop have to be established. Other RC/AH systems with DLC [22] flight tested with TIFS, show that due to high DLC effectiveness close coupling of pitch/heave motion degrade pilot ratings.

7. Due to system characteristics of RC/AH-system the pilot is much more sensitive to the initial and steady state rate response due to command inputs than for conventional aircraft. By freeing the pilot from stabilization task he has time enough to pay attention to such details.

8. New handling qualities criteria for RC/AH must take the following factors into account: a) single-loop flight path control situation, b) modified pilot techniques and c) intermittent entry of pilot into the loop.

6. CONCLUSIONS

As shown in the previous discussion the following general conclusions can be drawn.

1. The short-period-frequency requirements of MIL-F-8785 C are not adequate for precise flight path control tasks using advanced control systems with blended DLC.

2. It is necessary to separate the requirements for pitch and for heave motion.

3. For conventional flight path control techniques using pitch attitude, it has been shown that, the flight path to pitch coupling behaviour is an important handling qualities parameter.

4. The flight path to pitch attitude phase at the frequency of pilot closed inner pitch loop is proposed as criterion parameter in representing the heave/pitch control harmony or flight path to pitch loop separation.

5. The γ/θ-phase criterion is suitable for conventional and DLC-enhanced aircraft. A simple conversion to the $\omega_{nsp}^2/n/\alpha$-criterion is possible if n/α is interpreted as $-Z_\alpha/g$.

6. As shown by the RC/AH-system results, flying qualities have to be system oriented also. Control augmentation will change pilot task and pilot-aircraft control situation; therefore, special handling qualities requirements are necessary.

7. It is questionable whether a closed loop system representation is applicable to RC/AH-systems. The pilot exhibits discrete control behaviour and open-loop-type control techniques. Further investiations are required to take these factors into account for future handling qualities requirements.

7. REFERENCES

1. Hamel, P., Wilhelm, K.,Hanke, D.,Lange, H.-H., "Steep approach flight test results of a business-type aircraft with direct lift control", AGARD CP 160, 1974.

2. Hanke, D., "In-Flight investigation of direct lift control in landing approach", DFVLR IB 154-77/25, Aug. 1977.

3. v. Gool, M.F.C., Hanke, D., Lange, H.-H., "Flight path tracking experiments in the DFVLR HFB 320 equipped with direct lift control", DFVLR IB 154-75/32, 1975.

4. Hanke, D., Lange, H.-H., "In-flight handling qualities investigation of various longitudinal short term dynamics and direct lift control combinations for flight path tracking using DFVLR HFB 320 variable stability aircraft", AGARD CP 260, 1978.

5. Schafranek, D. "Theoretische Untersuchung über den Einfluß der Spoilerdynamik auf die Fliegbarkeit eines Flugzeuges mit direkter Auftriebssteuerung" (Theoretical investiation of the effect of spoiler dynamics on the handling of an aircraft with direct lift control), DFVLR-FB 80-07 1980, ESA-TT-681.

6. Hanke, D., "In-Flight Handling Investigation of A 310 Airbus DLC-Configurations on Landing Approach, Using the DFVLR-HFB 320 In-Flight Simulator (Variable Stability Aircraft). DFVLR-FB 79-18, Feb. 1979, ESA-TT-630, Nov. 1980.

7. Lange, H.-H., Hanke, D., "Use of DFVLR In-Flight Simulator HFB 320 Hansa for Handling Qualities Investigations, 12th ICAS-Congress, München, 12.-17.10.1980, Preprint ICA-80-9.4.

8. Anon,"MIL-F-8785 C, Military Specification-Flying qualities of piloted airplanes."

9. Hoh, R. H. et al., "Development of handling qualities criteria for aircraft with independent control of six degrees of freedom," AFWAL-TR-81-3027, April 1981.

10. Bihrle, W. jr., "A handling qualities theory for precise flight path control", AFFDL-TR-65-198, June 1966.

11. Hodgkinson, J., Berger, R.L., Bear, R.L., "Analysis of high order aircraft/flight control system dynamics using an equivalent system approach", McDonnel Aircraft Company, MCAIR 76-009, 1976.

12. A'Harrah, R. C., Lamanna, W. J., Hodgkinson, J., "Are todays specifications appropriate for tomorrow's airplanes?, AGARD CP 260, Sept. 1978.

13. A'Harrah, R. C., Woodcock, R. J., "The military flying qualities specification, a help or hindrance to good fighter design?, Preprint AGARD FMP Symposium, Florence, Italy, Oct. 1981.

14. Neal, T. P., Smith, R. E., "An in-flight investigation to develop control system design criteria for fighter airplanes",AFFDL-TR-70-44 Vols. I and II, Air Force Flight Dynamics Laboratory, Wright-Patterson AFB, Ohio, Dec. 1970.

15. Mooij, H. A., Boer, W. P., van Gool, M.F.C., "Determination of low-speed longitudinal manoeuvring criteria for transport aircraft with advanced flight control systems", NLR TR 79125, Dec. 1979.

16. Ashkenas, I. L., "Summary and interpretation of recent longitudinal flying qualities results", Journal of aircraft 8 (1971), Vol. 5, p. 324-328.

17. Stapleford, R. L.,Ashkenas, I. L., "Effects of manual altitude control and other factors on short-period handling quality requirements", Journal of aircraft 5 (1968), Vol. 1, p. 41-48.

18. Sudderth, R. W., "Development of longitudinal handling qualities criteria for large advanced supersonic aircraft", Final report, NASA CR-137635, 1975.

19. Neal, T. P., Chalk, C. R., "Revision of military specification MIL-F-8785 B (ASG), flying qualities of piloted airplanes", Cornell Aeronautical Lab., Flight Research Dept. Cal FRM No. 462, 1971.

20. Ashkenas, I. L. et al., "Recommended revisions to selected portions of MIL-F-8785 B and background data", STI prepared for AFFDL, AFFDL-TR-73-76, August 1973.

21. Stapleford, R. L., Klein, R. H., Hoh, R. H., "Handling qualities criteria for the space shuttle orbiter during terminal phases of flight", NASA CR-2017, 1972.

22. Weingarten, N. C., "Flying qualities Experiment performed for the National Aerospace Laboratory (NLR), The Netherlands,in the total in-flight simulator (TIFS)", Calspan Report No. 6645 - F-4, March 1981.

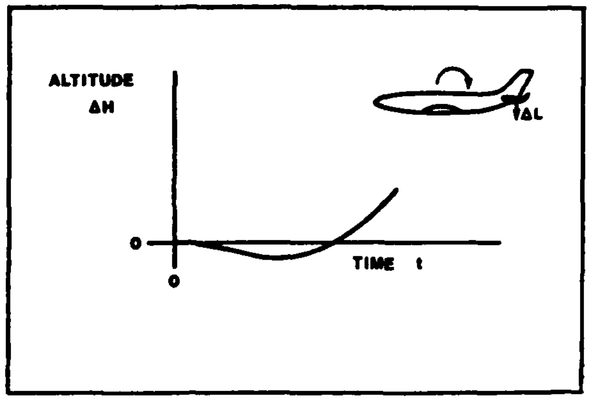


Fig. 1 Altitude response due to elevator input

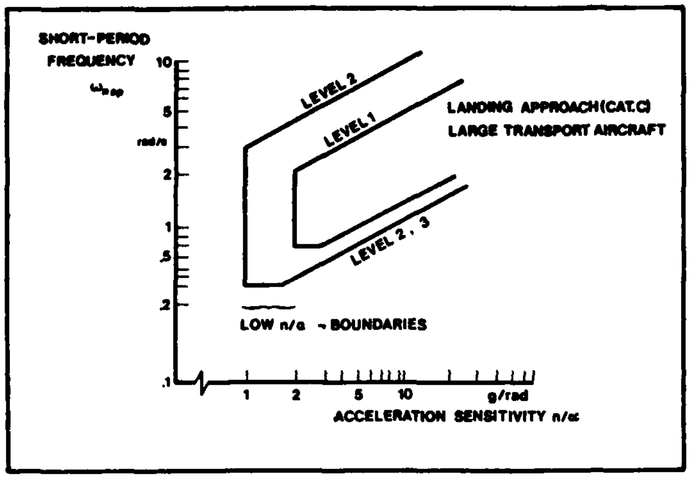


Fig. 2 Low n/α-requirements of MIL-F-8785 C

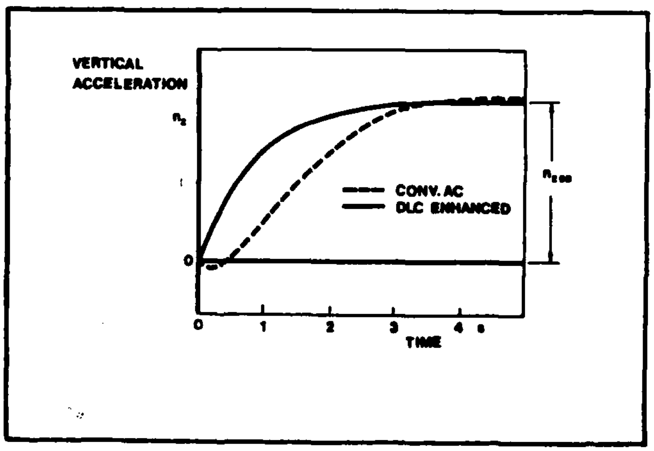


Fig. 3 Short-term vertical acceleration response of conventional and DLC-enhanced aircraft

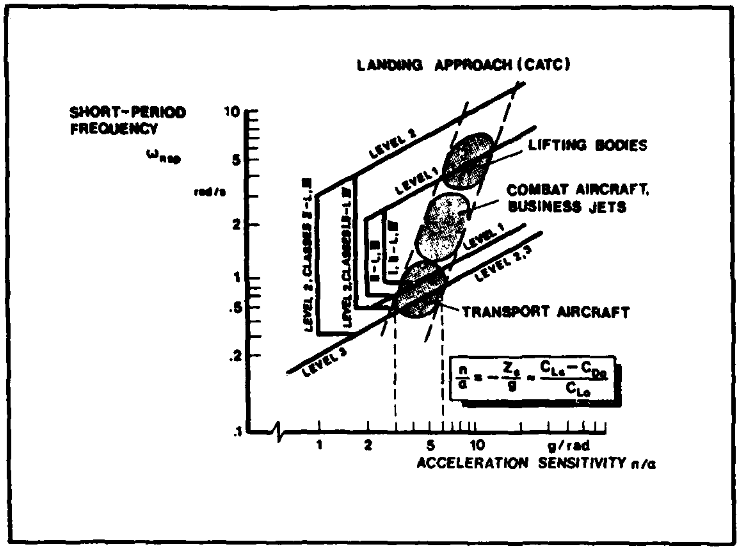


Fig. 4 Comparison of existing aircraft with MIL-F-8785 C CAT. C requirements

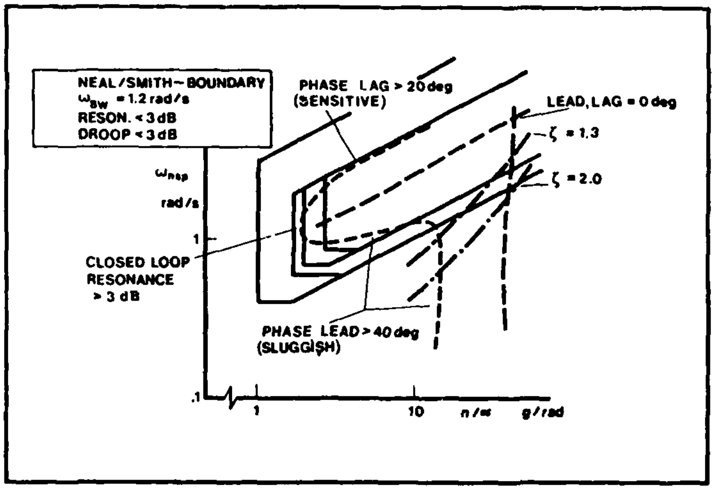


Fig. 5 Comparison of MIL-F-8785 C CAT. C requirements with Neal-Smith closed loop criterion

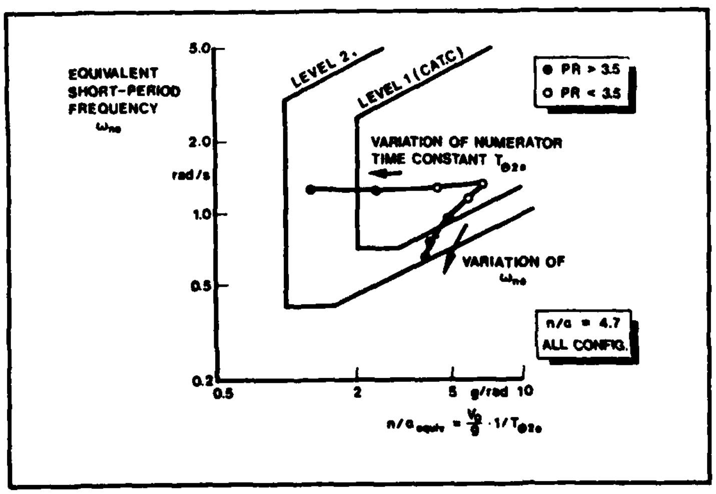


Fig. 6 Comparison of independent $1/T_{\theta 2}$ and ω_n variation with MIL-F-8785 C CAT. C requirements. Data from [15]

Fig. 7 DFVLR-HFB 320 In-flight simulator

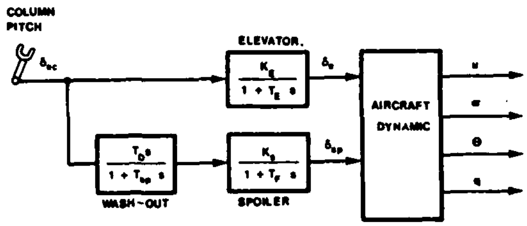


Fig. 8 Schematic of open loop direct lift control system

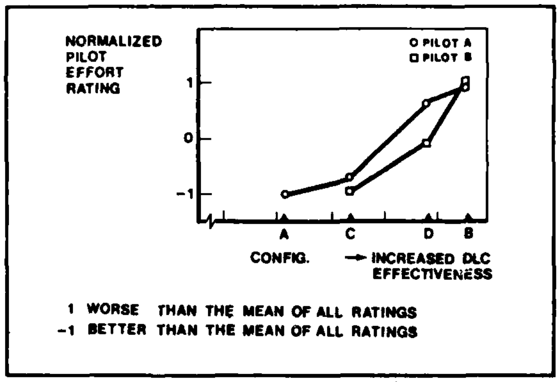


Fig. 9 Normalized pilot effort ratings

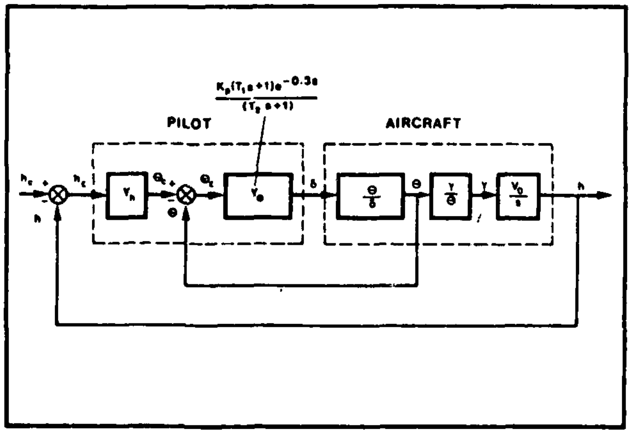


Fig. 10 Manual altitude hold situation during landing approach

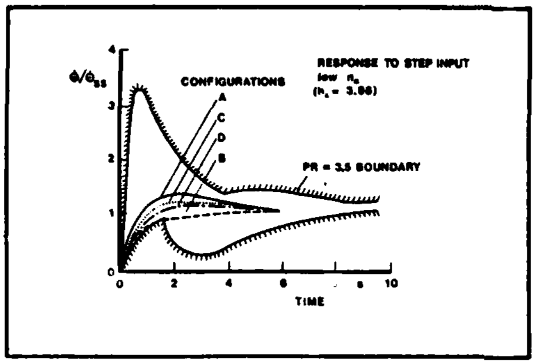


Fig. 11 Pitch rate time response criterion for low speeds [18]

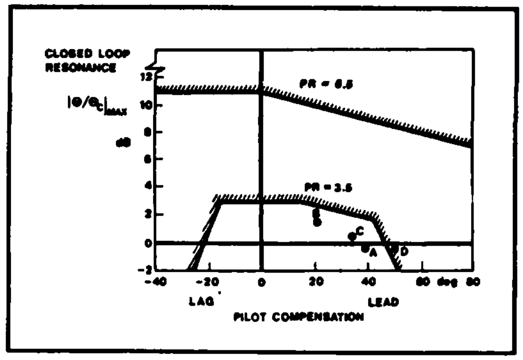


Fig. 12 Comparison of DLC-Configurations with the Neal-Smith Criterion

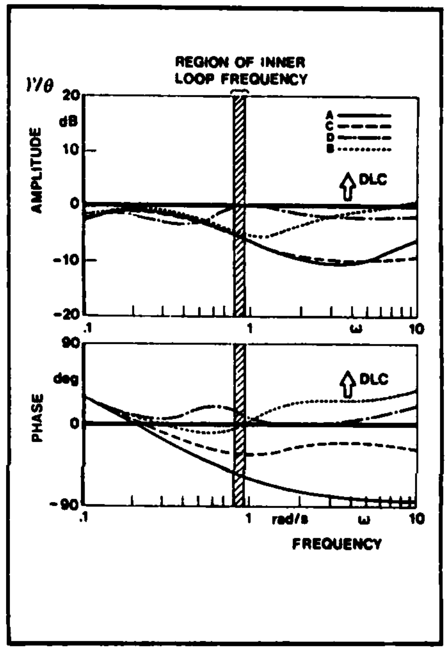


Fig. 13 Influence of DLC on γ/θ-transfer function

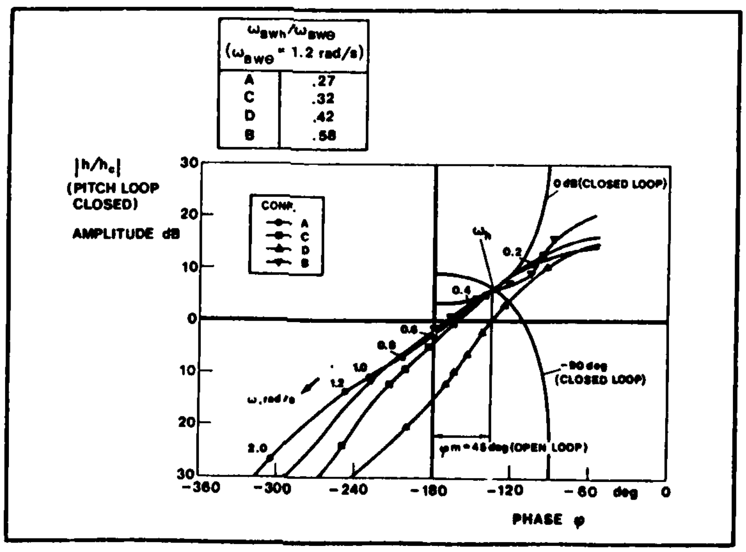


Fig. 14 Nicholsdiagram of h/h_c-transferfunction (inner pitch loop closed by the pilot)

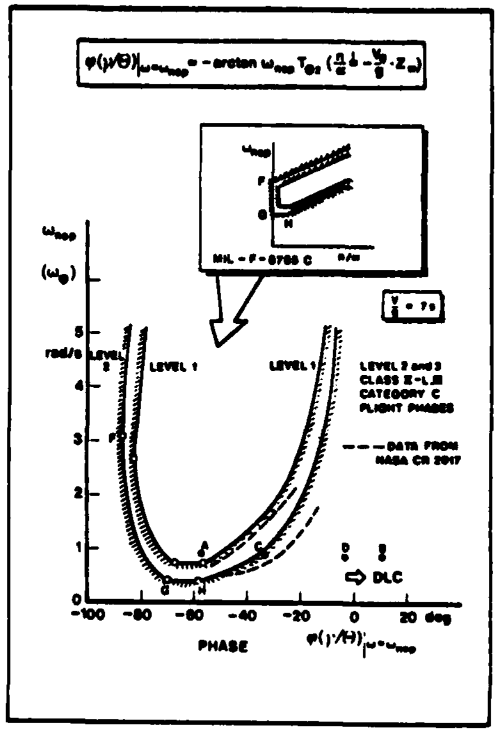


Fig. 15 Conversation of MIL-F-8785 C CAT. C requirements into the γ/θ-Phase diagram

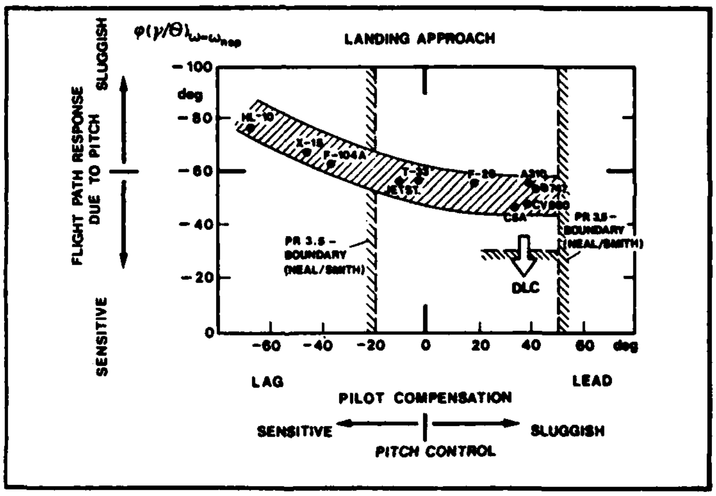


Fig. 16 Combined flight path/pitch representation using γ/θ-Phase and Neal-Smith criteria

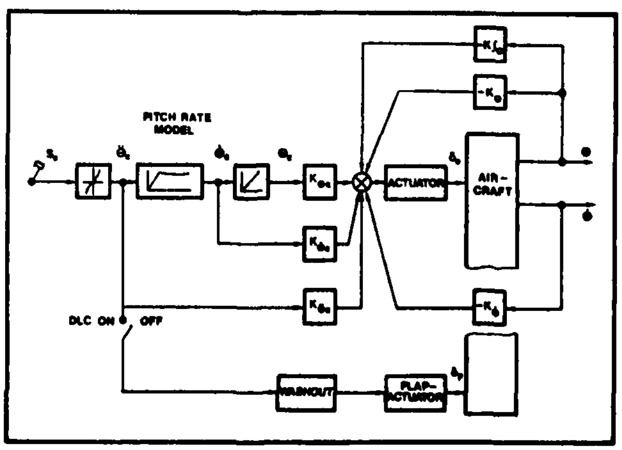


Fig. 17 Schematic of rate command/attitude hold system with DLC (pitch axis)

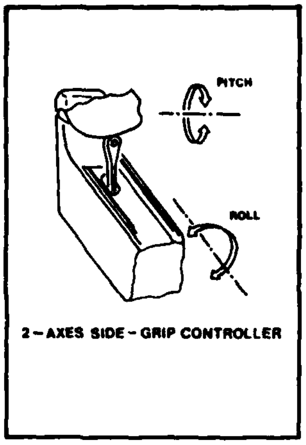


Fig. 18 2-axes sidegrip controller

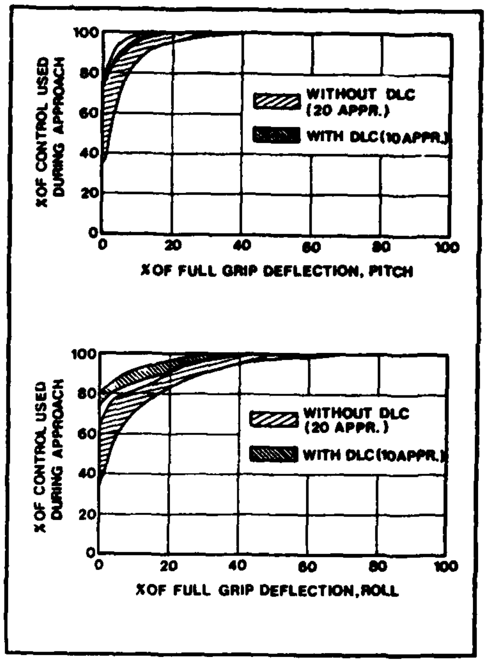


Fig. 20 Usage of sidegrip related to approach time with and without DLC (RC/AH-System)

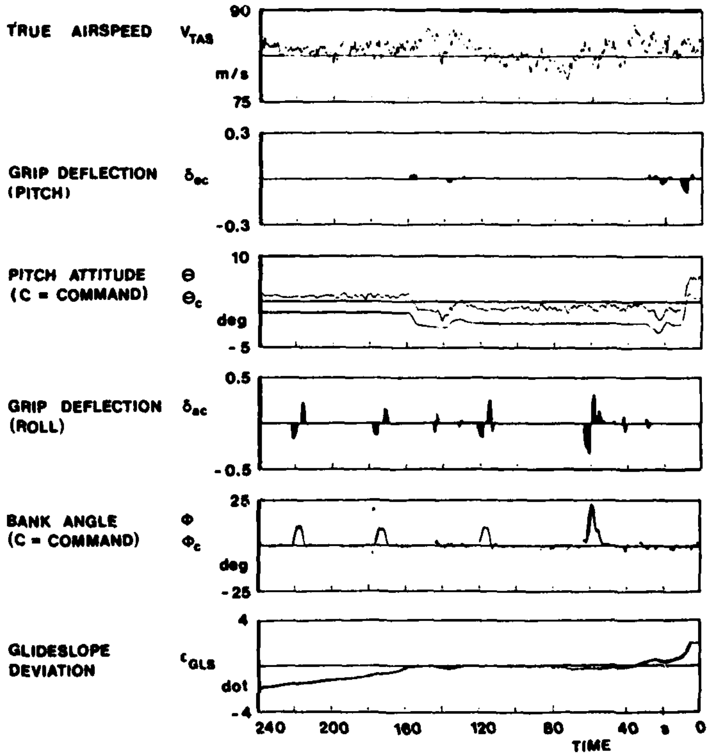


Fig. 19 Landing approach under gust conditions with RC/AH-System

THE STATUS OF MILITARY HELICOPTER HANDLING QUALITIES CRITERIA

David L. Key
Chief, Flight Control Division
Aeromechanics Laboratory
U.S. Army Research and Technology Laboratories (AVRADCOM)
Ames Research Center, Moffett Field, California 94035

SUMMARY

The paper provides an assessment of current helicopter specifications and describes the plans that the U.S. Army AVRADCOM has for a major effort to develop a new specification providing mission-oriented requirements.

Comparisons of several previous U.S. military specifications indicate that MIL-F-83300 has clear advantages in its broad coverage of important handling qualities aspects and its systematic structure. Its disadvantages are that it is primarily based on V/STOL data, and explicit helicopter characteristics are only lightly covered. MIL-H-8501A and the system specifications developed expressly for recent military procurements do specifically address helicopters, and, through long familiarity, the helicopter community is comfortable with them. However, they do have rather sparse coverage of many important topics. All of these specifications are sadly lacking in mission-oriented criteria and are basically for visual meteorological conditions (VMC) with no coverage of night operations and only token requirements for instrument meteorological conditions (IMC).

Recognition of these deficiencies has resulted in a major effort being initiated by the U.S. Army and Navy to develop a new specification containing mission-oriented handling qualities requirements. The efforts will be directed by the U.S. Army AVRADCOM, and contributions to the program will be made by NASA, the USAF, and the FAA. It is planned that a revised specification will be ready for submission for adoption as MIL-H-8501B in 1985.

1. INTRODUCTION

The current specification MIL-H-8501A, Helicopter and Ground Handling Qualities (Ref. 1), is a 1961 revision of a 1952 document. It gave good guidance in its early years, but by the late 1960's had many obvious deficiencies. For example, Ref. 2 provides a detailed analytical review of these shortcomings; empirical evidence can be seen in reports of flight-test evaluations by the Army Engineering Flight Activity (AEFA) (Refs. 3-5). MIL-H-8501A is still used by AEFA, to some extent, as a yardstick for flight-test evaluation, but for procurement of the UTTAS and AAH, the Army Aviation Research and Development Command (AVRADCOM) developed a new set of handling qualities specifications and incorporated them into the Prime Item Development Specification (PIDS) (Refs. 6,7). The Navy used essentially the same requirements for the LAMPS-III. For the Army Helicopter Improvement Program (AHIP), the handling quality requirements referred to MIL-H-8501A and provided some guidance on control sensitivity and rate damping (Ref. 8).

There have been several formal attempts to revise MIL-H-8501A – a "B" version was proposed in 1968 but never developed and adopted. The V/STOL specification MIL-F-83300 (Ref. 9) was the culmination of a major effort by Cornell Aeronautical Laboratory under the sponsorship of the Air Force Flight Dynamics Laboratory. It incorporated all the data available at the time and followed closely the structure and format of the recently revised specification for conventional aircraft – MIL-F-8785B (Ref. 10). The data and rationale for the requirements were presented in a background information and users guide (BIUG)(Ref. 11) which was modeled after the equivalent BIUG for MIL-F-8785B (Ref. 12). MIL-F-83300 attempted to include helicopters and, in fact, was adopted for helicopter application by the USAF. However, the U.S. Navy and Army chose not to adopt it for helicopter application. Some of the reasons for this may be related to the type of criticisms provided by Green (Ref. 13). In an attempt to overcome the perceived shortcomings of MIL-F-83300, the Army and Navy jointly sponsored Pacer Systems, Inc., to draft a revision to MIL-H-8501A. This effort adopted some of the concepts and structure used in MIL-F-8785B and 83300, and many of the new requirements were innovative, though they lacked data for substantiation. The preliminary report of this effort was submitted in March 1973; it had limited distribution, and was never finally published.

Experience in previous efforts to revise MIL-H-8501A showed that the primary obstacle to developing new requirements was a lack of systematic data from which new criteria could be developed and used for substantiation. In the last 10 years, several sources have contributed toward enlarging this technical data base: significant experience has been gained in procurement of three Army projects, the UTTAS (UH-60A Blackhawk), the AAH (AH-64 Apache), and the AHIP Near-Term Scout Helicopter (NTSH); the Navy has procured the LAMPS III (Seahawk), which is based on the Blackhawk; experimental research

work specifically oriented toward building the flying qualities data base has been under way by the Army and NASA at Ames Research Center; significant strides have been made by the fixed-wing community toward developing techniques for analysis and understanding of flying qualities, particularly techniques to handle fly-by-wire digital control systems, and tailored responses for the integration of flight control and weapon delivery maneuvers exploiting features such as direct force control. This body of experience will form a reasonable basis for mounting a major revision effort.

The remainder of the paper is divided into three main sections. The first provides an overview of existing specifications and describes some of the basic needs. The second illustrates some of the problems with two of the most fundamental criteria: longitudinal dynamic response and roll-control effectiveness. The last section outlines plans for the joint Army/Navy program to develop a new specification incorporating mission-oriented criteria which will be called "Update 8501."

2. OVERVIEW OF CURRENT SPECIFICATIONS

Reference 14 describes an attempt to assess the current specifications by comparing the latest UTTAS and AAH system specifications (Refs. 15,16) with MIL-H-8501A (Ref. 1) and MIL-F-83300 (Ref. 14). Three other specifications, AGARD Reports 408 and 577-70 (Refs. 17,18) and Curry and Matthews (Ref. 19) made notable contributions to the development of V/STOL criteria, but they were not included in this comparison because they made no claims to cover helicopters, and were not written as contractual documents. The comparison shows that MIL-F-83300 has clear advantages in its broad coverage of important handling qualities aspects and its systematic structure. Its disadvantages are that it is primarily based on V/STOL data, and explicit helicopter characteristics are only lightly covered. MIL-H-8501A and the PIDS do specifically address helicopters, and, through long familiarity, the helicopter community is comfortable with them. However, they do have rather sparse coverage of many important topics. Even where topics are addressed, many shortcomings in the MIL-H-8501A requirements have long been recognized (Ref. 2). In addition, MIL-H-8501A and the PIDS lack a systematic treatment of flight envelopes and failures. All of these specifications lack mission-oriented criteria and are basically for VMC with only token recognition of separate IMC requirements.

It must be recognized that the task or mission flight phase to be performed by the helicopter can have a substantial effect on the requirements. In MIL-F-83300 the requirements were divided into hover and low-speed (i.e., less than 35 knots) and forward flight (i.e., for speeds from 35 knots to V_{CON}). The data base for the hover and low-speed requirements were based largely on research investigations using generalized hover and low-speed taxi tasks. No systematic investigations had been made of mission-oriented tasks, such as night NOE flying where precision of control is required in tasks such as hover-bobup (Ref. 20), or shipboard landings in high sea state, where the ship motions and the wind and turbulence interactions result in an extremely taxing task (Ref. 21). Attention to such flight phases will necessitate significantly more stringent requirements. For speeds greater than 35 knots the data and requirements were oriented at V/STOL approach and landing. The resulting requirements may be good minimums for flight safety but there is need to develop requirements to enhance performance of the operational missions that helicopters perform in this speed range. The intent in MIL-F-83300 was to convert to the fixed-wing aircraft requirements of MIL-F-8785B at speeds above V_{CON}. Although the idea of making helicopters meet such rigorous requirements turned V"CONVERT" into V"CONSTERNATION," and was rejected, the advent of high-speed rotorcraft, such as the Bell XV-15 Tilt Rotor and the Sikorsky XH-59A ABC, means that this problem will soon have to be addressed.

Additional problems result from the need to perform increasingly complex missions in adverse weather and at night. The plethora of pilot aids for navigation, communications, weapons, survivability, and vision aids compete for attention and can add to the pilots' workload if not suitably integrated. Definition of meaningful handling-qualities criteria must therefore consider all of the pilot's tasks involved in the mission flight phase together with an integrated treatment of vehicle dynamics, flight control system characteristics, cockpit controllers, displays and vision aids.

In many ways helicopters are much more complex than fixed-wing aircraft. The inherent assymetry of single-rotor helicopters causes them to have several features that complicate analysis and specification of handling qualities criteria; there is a strong cross-coupling between longitudinal and lateral-directional responses, they are highly nonlinear, and they inherently involve more than the classical rigid-body modes used to represent the responses of conventional fixed-wing aircraft.

One form of cross-coupling manifests itself in the response to control (e.g., roll due to pitch control and pitch and roll due to collective control). These couplings cannot be eliminated by static considerations of control phasing since the cross-coupling is a function of the frequency content of the control input (Ref. 22). Another coupling phenomena, related to the engine governing system results in coupling between the yaw response and the rotor rpm control. This coupling is well described by Kucsynski et al. (Ref. 23). Briefly, the situation occurs as follows: the engine fuel-control governor senses a yaw rate as an rpm perturbation and tries to correct to the referenced value. This changes the rotor torque, which has to be reacted through the fuselage. Depending on the engine fuel-control system dynamics, the phasing can result in amplifying or damping the fuselage yaw response.

Nonlinearities occur both statically and dynamically. An example of static nonlinearity is the YAH-64 pitching moment due to sideslip. Increasing sideslip angles to left of trim required increasing aft longitudinal control to trim, and sideslip to the right required forward longitudinal control to trim. In addition, the trim required for right sideslip varied in a nonlinear fashion with both the sideslip angle and with the airspeed. This rather complicated pitch-to-sideslip coupling caused pilot coordination problems in turning maneuvering flight, and was determined to be a shortcoming (Ref. 24). An example of dynamic nonlinearity is discussed in the following section on longitudinal dynamics.

Current perceptions of the threat leads to the doctrine of flying close to the ground to take advantage of terrain masking. This is called nap-of-the-earth (NOE) flight. Since it is desired to perform this flight phase not just in day visual conditions but in poor weather and at night, some form of vision aid, such as a light intensification device, an infrared sensor, or a radar, will be required. Presentation of this out-of-the-window view for the pilot can be through a helmet-mounted device or through a panel-mounted display. In either case, it appears that symbology providing flight control information has to be provided to enhance the pilot's ability to perform his flying task. Aiken and Merrill (Ref. 20) describe an investigation of how the symbology and drive logic in a head-up format interact with the SCAS to influence the handling qualities. Such considerations also need incorporating into a new military helicopter flying qualities specification.

3. DISCUSSION OF TWO FUNDAMENTAL CRITERIA

3.1 Longitudinal Dynamics

Perhaps the most fundamental flying qualities requirement is longitudinal dynamic response in forward flight (i.e., speeds above about 35 knots). Consider first the various requirements from MIL-F-83300, MIL-H-8501A, and the UTTAS and AAH PIDS that seem to be pertinent to this topic. Where these criteria can be transferred into graphic form, they have been plotted on the MIL-F-83300 boundary of $\omega_n \sim 2\zeta\omega_n$ (Fig. 1). The MIL-H-8501A requirement for n_z to be concave down within 2 sec has been plotted using the analysis provided by Seckel (Ref. 25).

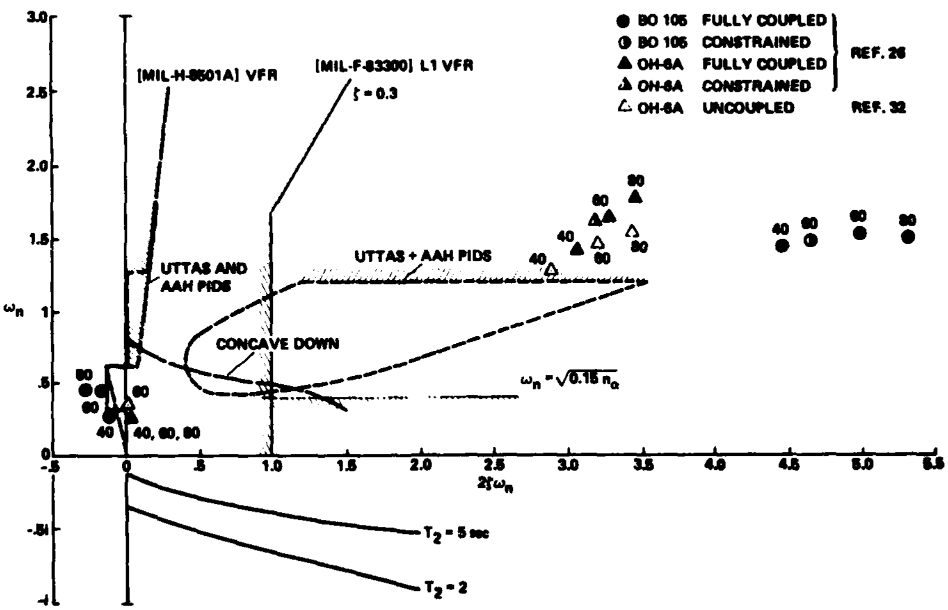


Fig. 1. Longitudinal dynamic requirements.

The most obvious point for comment is that the MIL-H-8501A boundary is meant to cover all oscillatory modes throughout the frequency range. This leaves the critical short-term response essentially unspecified. MIL-F-83300 concentrates on the short-term response and adds an outer limit that all other roots must be stable. The two PIDS provide a low-frequency boundary similar to, but not exactly the same as, the MIL-H-8501A VFR boundary, but do add a short period "bullseye." Superimposed on the figure are some root locations for the OH-6A and BO-105 helicopters taken from Ref. 26. The low-frequency roots of the BO-105 are unstable and do not meet any of the sketched requirements. However, as discussed in a paper by Pausder and Jordan (Ref. 27) and demonstrated in many years of successful application, the BO-105 is indeed a satisfactory VFR aircraft. Neither the BO-105 nor the OH-6A meets the "bullseye" required by the UTTAS and AAH PIDS. It is not at all clear to what extent this would make for an unacceptable response.

Because of special helicopter idiosyncrasies, there are more subtle questions that need to be addressed. First, consider the effect of cross-coupling. Helicopters have significant cross-coupling between longitudinal and lateral directional responses so that an arbitrary decoupling of longitudinal from lateral directional degrees of freedom may or may not be valid when comparing roots for the boundaries. Both MIL-F-83300 and MIL-H-8501A are ambiguous on this point, but the UTTAS and AAH PIDS both state that the boundary refers to motions with controls fixed following the disturbance. This implies that the longitudinal motions are being addressed with full coupling from whatever lateral directional perturbations are developed. Heffley et al. (Ref. 26) performed an analysis to determine the effect on dynamics of one axis while regulating the off-axis motions with a simple pilot model. The fully coupled and fully constrained roots at 60 knots are shown for the OH-6A and BO-105 in Fig. 1. Also shown are roots computed for the OH-6A by using just the longitudinal equations. For the high-frequency modes there is a noticeable difference; however, it is probably not significant. For the low-frequency (phugoid) roots there is a noticeable difference for the OH-6A but not the BO-105. Since these low-frequency roots are very close to the boundary, a difference of the magnitude shown for the OH-6A could result in being on the wrong side of the boundary and may be significant for interpreting the requirements.

The second property of helicopter dynamics which must be considered is nonlinearity. Tomlinson and Padfield (Ref. 28) describe results of a ground-based simulation to investigate helicopter agility. They describe nonlinearities in pitch response that are quite significant with stiff hingeless rotors. The nonlinear effects are due to airspeed changes that start to occur only a few seconds after the control input. Figure 2 is taken from Ref. 28; it shows a time history of normal acceleration and pitch rate for two rotors, a very stiff rotor, D2 (equivalent hinge offset based on flapping frequency e ≈ 18%) and a much more flexible rotor, A2 (e ≈ 3%). The vertical scale is normalized by the size of pitch input and scaled to the largest input case. Although there are negligible differences in response character with the soft rotor, with the stiff rotor the character changes dramatically after about 1.5 sec, a sharper peak occurring much earlier with the larger inputs. Also, for the stiff rotor, the larger the input, the more the response departs from being a rate type, and pilot appreciation of pitch control would be expected to change. Linear theory will, of course, predict the type of response produced by the smaller inputs; hence, handling-qualities criteria based on such analytical methods may be at variance with pilot opinion of handling qualities during maneuvering tasks.

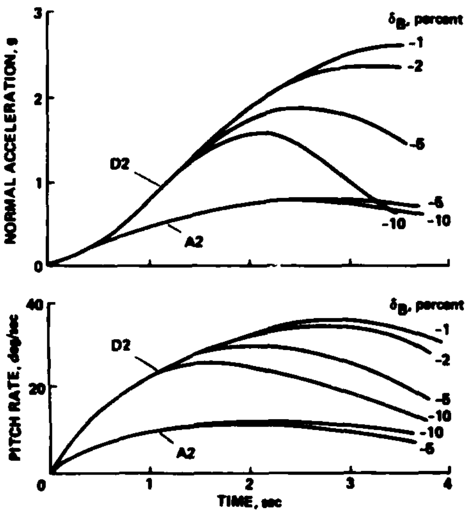


Fig. 2. Response to different levels of pitch control (from Ref. 28).

In addition to these peculiarly helicopter problems, there are many questions common to the fixed-wing community that must be considered; for example, how to specify the level of augmentation required, and provide guidelines for control system synthesis, especially when forward loop-shaping is included so that the response to control is different from the response to disturbances.

3.2 Roll-Control Effectiveness

Another extremely important and fundamental criteria which can have an effect on the basic design of the helicopter is roll-control power or control effectiveness. Again, the pertinent paragraphs from the four specifications – MIL-F-83300, MIL-H-8501A, and the UTTAS and AAH PIDS – will be considered.

MIL-F-83300 specifies roll-control effectiveness in terms of the time to bank 30°. The numbers were based on data generated for MIL-F-8785B and correspond to the flight phase Category C (terminal flight phases such as approach, landing, and takeoff) for small and medium-sized aircraft. This reflects the V/STOL emphasis in MIL-F-83300, since V/STOLs, other than helicopters, are in the flight regime of interest only during landing and takeoff. Helicopters, of course, perform most of their functions in this speed regime and can require controllability quite different from simply landing or taking off.

MIL-H-8501A requires that at least 10% of the maximum attainable hovering control effectiveness be available at all loading and flight conditions. The hover criteria specify a minimum bank angle to be achieved in 0.5 sec. The bank angle change is a function of weight; it is plotted in Fig. 3. Specifying the bank-angle change in such a short time (0.5 sec) puts an overemphasis on the effect of lags and delays in the control system and is very sensitive to the form of input, that is, the approximation to a perfect step. A NASA study (Ref. 29) showed that 0.1-sec transport delay followed by 0.3-sec ramp is a reasonable approximation to pilot input. After 1 sec, such an input provides about 60% of the response that would be achieved following a perfect step. At 0.5 sec the difference would be even greater.

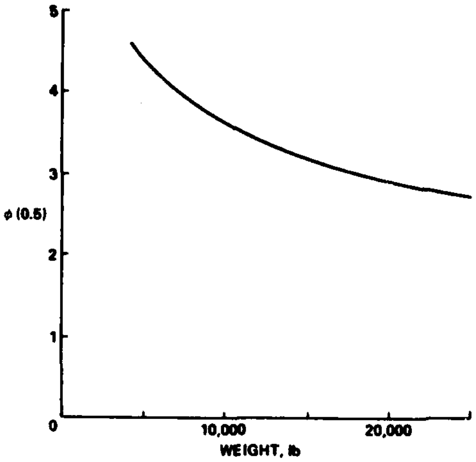


Fig. 3. MIL-H-8501A hover roll control effectiveness requirement.

The UTTAS PIDS has no explicit roll-effectiveness criteria. It requires gearing and delays to be satisfactory, and by virtue of the pullup-pushover maneuver implies that some roll-control power may be required at 0 g. This assumption may not be true, because although the use of control to maintain bank angle within limits is not explicitly required or forbidden for the UTTAS, for the AAH, the same bank-angle limits are imposed during the pullup-pushover maneuver, but this time to be achieved with roll-control fixed.

The AAH PIDS defines a minimum gearing or sensitivity value (13°/sec/in.) and demands linearity. If the control travel is the typical ±3 to 5 in., linearity would achieve at least 40°/sec. "Response time" is required to be 0.7 to 1.1 sec. Response time probably refers to the roll time-constant; if so, the numbers are questionable, since the BO-105 and OH-6A have roll time-constants of 0.1 and 0.2 sec, respectively.

The MIL-H-8501A requirement that 10% hover control effectiveness be available for overcoming gusts and upsets and for maneuvering would seem to be insufficient. However, despite past pleas for documented control usage (Ref. 30), there is, unfortunately, still little hard data on exactly what is required. Experimenters occasionally specify the maximum available control power, for example, in terms of $L_{\delta A}\ \delta_{AMAX}$, but seldom give data on how much was actually used during the maneuvers investigated. An ideal form of data presentation is probability density plots, but this format is cumbersome and there is the question of how much the low usage tails can be cut off. A recent X-22A report (Ref. 31) shows that the maximum control really needed by the pilot correlated very well with the 3 sigma value.

Clearly, mission flight phase has a large effect on the control power required to manuever. The study by Tomlinson and Padfield (Ref. 28) referred to earlier showed that response to roll control for the configurations investigated was typically set at a value that allowed a 30° bank angle to be achieved in 1.5 sec with 1.0 in. of control deflection. Control travel was ±5 in.; extrapolating to full control input shows that time to bank 30° would be between 0.5 and 0.7 sec for the more linear configurations. One configuration required as long as 2.4 sec to bank 30° with 1.0 in. of stick, and this was rated as having barely sufficient control.

Figure 4 shows a plot of control effectiveness boundaries on a scale of control power $L_{\delta A}\ \delta_{AMAX}$, and control sensitivity $L_{\delta A}$, versus roll time-constant τ_R. This modified plot was taken from the MIL-F-83300 BIUG (Ref. 11), for which the original plot was obtained from a Princeton (Ref. 33) simulation of carrier landing approach. The t_{30} lines were computed using a single degree of freedom approximation, and a 0.2-sec ramp control input. Plotted on the figure are the $L_{\delta A}\ \delta_{AMAX} \sim \tau$ values for the BO-105 and OH-6A (constant for the speed range 40 to 80 knots; taken from Ref. 26). Also shown is the range of "response time" that is specified in the AAH PIDS. Clearly, much work needs to be done toward specifying this important parameter for helicopter missions.

4. PLANS FOR UPDATE 8501

The Army and Navy have initiated a systematic effort to develop a new general specification for the handling qualities of military rotorcraft. The effort will build upon the ideas, techniques, and technology developed by the fixed-wing community, as well as utilize the available experience with current helicopter specifications and V/STOL criteria. The existing data base will be used to the maximum extent possible and supplemented by new data obtained under the auspices of this and related projects. Specific programs developing new data bases for this purpose are being performed by the Army Aeromechanics Laboratory and by NASA at Ames Research Center, and by the Navy at NADC, Warminster. The primary effort of developing the specification revision will be performed under contract, proposals for which are currently under evaluation; a contract award is planned for August 1982.

Rotorcraft to be covered must include conventional single and tandem rotor-helicopters and, to the maximum extent possible, high-speed rotorcraft such as compound (e.g., Lockheed AH-56A) and novel configurations (e.g., Bell XV-15 Tilt Rotor and the Sikorsky XH-59A ABC). This is a joint Army/Navy program, and the resulting specification must address the mission requirements of both services. The contracted effort will be directed by the Army Aviation Research and Development Command (AVRADCOM), technical responsibility being shared between the Aeromechanics (AL)(Research and Technology Laboratories) and the Directorate for Development and Qualification (D and Q). Contributions to the program will also be made by NASA, the USAF and the FAA; a Technical Committee has been formed to coordinate all these inputs.

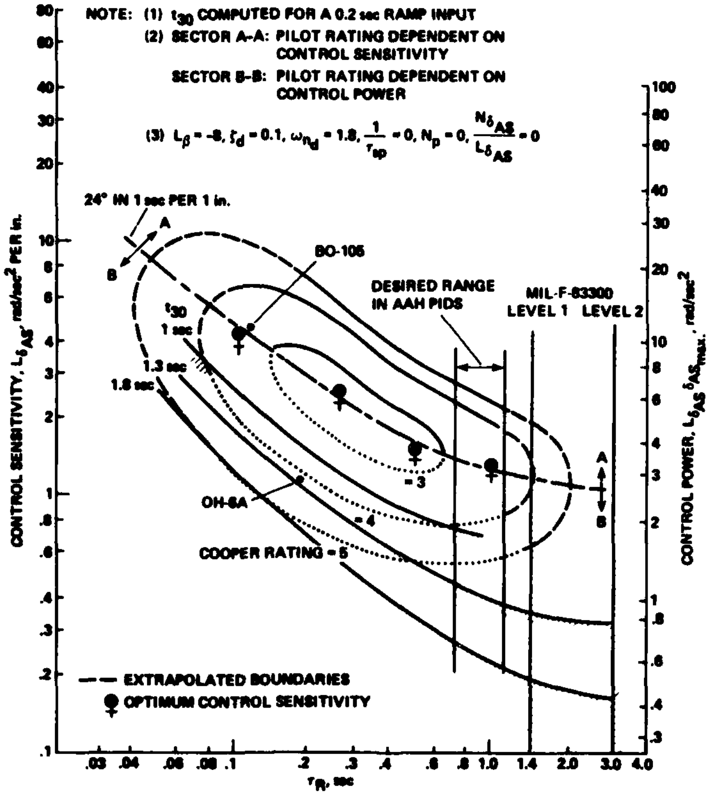


Fig. 4. Roll control effectiveness boundaries (from Refs. 11 and 33).

Contractor efforts will consist of analysis and evaluation. Access will be provided to Army/Navy (Marine) aviation users for mission task analysis and possibly for flight test and simulation demonstration and evaluation.

The contracted effort will be divided into two Phases: Phase 1 will involve developing a specification structure, incorporating existing data-base/criteria into that structure, and defining critical gaps and possible ways for addressing these gaps. Phase 2 will extend the effort by finalizing the structure, incorporating any new data and criteria, and producing a proposed specification and background document. This will then be distributed for government and industry review, and the comments will be reworked into a final military specification and BIUG that can be submitted for adoption.

The proposed schedule is shown in Fig. 5. In phase one the contractor will perform studies and analysis to develop a specification structure that adequately addresses the following considerations:

1. Rotorcraft types and roles, including high-speed rotorcraft and compounds.

2. Levels of flying qualities

3. Flight envelopes

4. Systematic treatment of failures and reliability

5. Categorization of mission flight phases

6. Flight phase environment, including day, night, visibility, altitude, terrain nature, and atmospheric disturbances

7. Controllers (including side-stick controllers), displays and vision aids

Missions for Army/Navy rotorcraft will be analyzed to determine and characterize those flight phases which may have a design impact. Army missions will include those of the Scout, Attack, Utility, and Cargo types; Navy missions will include ASW, Marine Attack/Assault, Vertical Replenishment, and Ship-to-Shore Cargo, with particular emphasis on shipboard launch and recovery. A method must be developed and applied to define and quantify these flight phases and their constituent tasks. The form developed must be useful for handling qualities criteria development and definition, and must be compatible with the proposed specification structure.

It will be necessary to develop and use analytical techniques such as closed-loop analysis, pilot model analysis, and equivalent system representation to characterize the pilot, the rotorcraft and the pilot/rotorcraft interface. These characterizations must

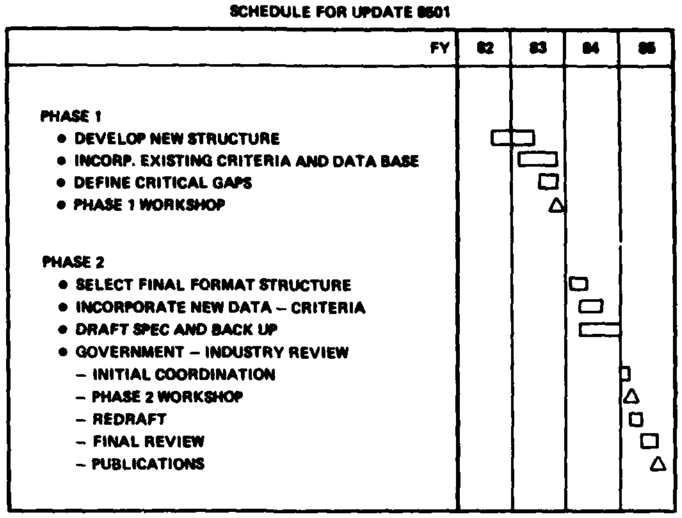


Fig. 5. Schedule for "Update 8501."

be capable of quantitative representation of the pilot/rotorcraft while performing the flight phase tasks of interest. The rotorcraft representation will include higher order and nonlinear effects, such as rotor dynamics, control system modeling, engine-rotor interactions, and interaxis coupling, with a sufficient level of detail to address effects which, for the flight phase tasks being considered, could influence the pilot's workload or task performance. Some important topics will no doubt require further development to become useful tools in criteria development or specification (e.g., low altitude atmospheric disturbance models, terrain models and pilot models). These will be defined and prioritized in sufficient detail to allow definition of the problem areas for independent study efforts or research contracts.

The criteria in existing specifications will be reviewed to determine the extent to which they can be verified and supported utilizing the available data base. New or improved criteria will also be developed to the maximum extent possible. After defining what criteria are valid, and for what situations, they will be incorporated into the new structure. This effort must recognize the fact that different characteristics will be required for different mission flight phase tasks and mission environment conditions. Priority will be placed on defining criteria for circumscribing characteristics that may have a significant effect early in a helicopter design process. Where possible, Level 1, Level 2, and Level 3 boundaries will be defined (in this context, the Levels refer to Levels of Flying Qualities as defined in MIL-F-8785B)(Ref. 10). Once the existing data base has been used to the maximum extent, the topics not adequately covered will be defined and prioritized. Of high concern to the Army are criteria related to agility and maneuverability, especially in day conditions with relatively unlimited visibility, and to night precision tasks such as NOE hover, unmask, and weapon firing; the Navy is particularly concerned with shipboard launch and recovery. It is expected that significant shortcomings or complete voids will be found in the existing flying qualities data base. The contractor will be expected to outline experiments to generate new data to address some carefully selected critical issues.

The structure and criteria developed will be presented in the form of a draft military specification. In preparing the draft specification, the following guidelines will be observed:

1. The requirements should not inhibit the designer in choosing a design approach for satisfying the mission requirements.

2. The requirements shall be in such a form that some means of demonstrating compliance (e.g., through analysis, simulation, or flight test) can be defined.

3. The specification should be as complete as possible so that ideally it sets forth requirements that are individually necessary and that together form a sufficient set to assure the required level of flying qualities.

As part of the effort to revise MIL-F-8785 and in the development of MIL-F-83300, supplementary background documents were prepared which contained background information and guidance for the specification user, together with substantiation data for the individual requirements. These documents have been well received by the airframe companies and the government organizations responsible for applying the specification requirements, and similar documents will be required for the MIL-H-8501B effort.

5. CONCLUSIONS

Operational experience over many years suggests that current specifications provide guidance that will provide flight safety in benign VFR conditions and with modest flight tasks. Current missions involving operations in poor weather and at night, NOE flight while avoiding threats and possibly engaging in air-to-air combat, and extended conditions for shipboard launch and recovery, will require much more rigorous capabilities. New requirements need to be developed, and, to ensure that they do not get imposed unnecessarily, a format must be used that distinguishes those helicopters that must meet the requirement from those that do not need to.

Many operational missions require special displays and vision aids so these characteristics must be included in a flying-qualities specification along with the vehicle stability and control characteristics and controller characteristics.

Current specifications provide rather sparse coverage so that some topics are not addressed at all, and some very basic topics are covered rather poorly. For example, the longitudinal dynamic response requirement in MIL-H-8501A covers low-frequency oscillations but does not address the important short-term response. Boundaries have been added in the UTTAS and AAH PIDS to attempt to address this but the resulting boundary is questionable. No attempt whatsoever has yet been made to cover higher-order effects such as nonlinearities and cross-coupling. The requirement for roll-control effectiveness, which could have a major effect on the design of the rotor system, needs much more work to establish a data base for a substantial requirement.

A new program has been initiated by the Army and Navy to update the helicopter specification. The plan is to develop a new structure, develop and incorporate requirements to the extent that they can be substantiated and, after performing a coordination review with industry, promulgate a new specification. Undoubtedly, many topics will be identified that have not been adequately covered, so the critical gaps will be identified and given emphasis in the ongoing R&D data-base generation efforts.

6. REFERENCES

1. General Requirement for Helicopter Flying and Ground Handling Qualities. Military Specification MIL-H-8501A, 7 Sept. 1961.

2. Ashkenas, I. L.; and Walton, R. P.: Analytical Review of Military Helicopter Flying Qualities. Technical Report No. 143-1, Systems Technology, Inc., Hawthorne, Calif. Aug. 1967.

3. Bailes, E. E. et al.: Instrument Flight Rules Capability Evaluation CH-45B (TARHE) Helicopter. Final Report USAASTA Project No. 71-01, Dec. 1972.

4. Laing, E. J. et al.: Airworthiness and Flight Characteristics Test Production OH-58A Helicopter Unarmed and Armed with XM 27E1 Armament Subsystem. Final Report USAASTA Project No. 68-30 Oct. 1970.

5. Winn, A. L.: Instrument Flight Evaluation OH-6A Helicopter – Part I. Final Report USAASTA Project No. 72-06, Nov. 1973.

6. Systems Specification for UTTAS No. AMC-SS-2222-01000, Dec. 1971.

7. Systems Specification for Advanced Attack Helicopter, No. AMC-SS-AAH-10000A, July 1973.

8. AHIP Specification. Contract No. DAAK50-81-R-005, Attachment 2 of Request for Proposal.

9. Flying Qualities of Piloted V/STOL Aircraft. Military Specification MIL-F-83300, 31 Dec. 1970.

10. Flying Qualities of Piloted Airplanes. Military Specification MIL-F-8785B(ASG), 7 Aug. 1969.

11. Chalk, C. R. et al.: Background Information and User Guide for MIL-F-83300 – Military Specification – Flying Qualities of Piloted V/STOL Aircraft. AFFDL TR-70-88, Mar. 1971.

12. Chalk, C. R. et al.: Background Information and User Guide for MIL-F-8785B(ASG) – Military Specification – Flying Qualities of Piloted Airplanes. AFFDL-TR-69-72, Aug. 1969.

13. Green, D. L.: A Review of MIL-F-83300 for Helicopter Applications. American Helicopter Society Preprint 643, May 1972.

14. Key, D. L.: A Critique of Handling Qualities Specifications for U.S. Military Helicopters. AIAA Paper 80-1542, Aug. 1980.

15. Prime Item Development Specification for Utility Tactical Transport Aircraft System. Specification No. DARCOM-CP-2222-S1000C.

16. System Specification for Advanced Attack Helicopter, No. AMC-SS-AAH-H10000A, Oct. 1976.

17. Recommendations for V/STOL Handling Qualities. NATO AGARD Report 408, Oct. 1962.

18. V/STOL Handling. Part I: Criteria and Discussion; Part II: Documentation. NATO AGARD Report 577, June 1973.

19. Curry, P. R.; and Matthews, J. T.: Suggested Requirements for V/STOL Flying Qualities, USAAML TR 65-45 RTM 37, June 1965.

20. Aiken, E. W.; and Merrill, R. K.: Results of Simulator Investigation of Control System and Display Variations for an Attack Helicopter Mission. AHS Preprint 80-28, May 1980.

21. Proceedings of the Navy/NASA Flying Qualities Workshop held at Naval Postgraduate School, Monterey, Calif., Apr. 1977.

22. White, F.; and Blake, B. B.: An Improved Method of Predicting Helicopter Control Response and Gust Sensitivity. AHS Preprint 79-25, May 1979.

23. Kuczynski, W. A. et al.: The Influence of Engine/Fuel Control Design in Helicopter Dynamics and Handling Qualities. AHS Preprint 79-27, May 1979.

24. Stewart, R. L. et al.: Development Test One. Advanced Attack Helicopter Competitive Evaluation. Hughes YAH-64 Helicopter. Final Report USAAEFA. Project No. 74-07-2, Dec. 1976.

25. Seckel, E.: Stability and Control of Airplanes and Helicopters. Academic Press, NY and London, 1964.

26. Heffley, R. K. et al.: A Compilation and Analysis of Helicopter Handling Qualities Data. Vol. 1: Data Compilation, NASA CR-3144, 1979; Vol. 2: Data Analysis. NASA CR-3145, 1979.

27. Pausder, H-J; and Jordan, D.: Handling Qualities Evaluation of Helicopters with Different Stability and Control Characteristics. Paper No. 6, First European Rotorcraft and Powered Lift Aircraft Forum, Sept. 1975.

28. Tomlinson, B. N.; and Padfield, G. D.: Piloted Simulation Studies of Helicopter Agility. Vertica, vol. 4, 1980, pp. 79-106.

29. Innis, R. C.; Holzhauser, C. A.; and Quigley, H. C.: Airworthiness Considerations for STOL Aircraft. NASA TN D-5594, Jan. 1970.

30. Chalk, C. R.; and Key, D. L.: Experimental Design and Data Acquisition for VSTOL Flying Qualities Simulations. SAE Paper 690695, SAE National Aeronautics and Space Engineering Meeting, Oct. 1969.

31. Radford, R. C.; Andrisani, D.; and Beilman, J. L.: An Experimental Investigation of VTOL Flying Qualities Requirements for Shipboard Landings. NADC Report 77318-60, Aug. 1981.

32. Siegel, E.: Stability and Control Data Summary for Single-Rotor Helicopter, Hughes OH-6A. Hughes Helicopter Company Report No. 369-V-8010, Apr. 1975.

33. Seckel, E.; Miller, G. E.; and Nixon, W. B.: Lateral-Directional Flying Qualities for Power Approach. Princeton University Report No. 727, Sept. 1966.

L'IMPACT DE LA CAG (Commande Automatique Généralisée) (*)

SUR LES QUALITES DE VOL DES HELICOPTERES

Marcel KRETZ
Ingénieur en Chef
Giravions Dorand Industries
92151 Suresnes, France

RESUME

Au cours des années 1970 - 1980, un changement notable est intervenu dans les moyens envisagés pour contrôler les conditions de fonctionnement d'un rotor d'hélicoptère. S'inspirant par sa technique et sa philosophie de base de la CAG des voilures fixes, la nouveauté consiste à contrôler les phénomènes dynamiques (vibrations, instabilités) et aérodynamiques (décrochage, interactions, rafales) dans la bande des fréquences nettement plus élevées que la bande utilisée par le pilote automatique, couvrant les fréquences s'étendant jusqu'à 30 Hz au moins. Simultanément, l'évolution des hélicoptères militaires a créé des exigences nouvelles concernant les qualités de vol, et dont on peut trouver l'origine dans : l'augmentation de la charge sur le disque, l'arrivée des formules nouvelles des appareils à voilures tournantes (rotor basculant, ABC, X-Wing, rotor stoppable et escamotable) et l'élargissement du domaine de vol (μ plus élevé, Cz_{moy} plus élevé). La communication contient une analyse de l'impact que risque d'exercer la CAG sur les qualités de vol et une discussion des avantages que son introduction peut procurer. En conclusion, les limitations des commandes actuelles, provenant du plateau monocyclique, seront dans l'avenir repoussées grâce aux commandes nouvelles, orientées vers des systèmes optimisés, auto-adaptatifs à boucles multiples, permettant un degré supérieur d'optimisation des qualités de vol.

1. - INTRODUCTION

La dernière décade a vu un changement considérable en préparation dans le domaine de la commande des hélicoptères. Un effort, inspiré par la CAG des voilures fixes, est fait pour élargir la bande passante d'entrée afin de pouvoir contrôler les phénomènes dynamiques et aérodynamiques bien au-delà du domaine du pilotage monocyclique que le plateau cyclique classique autorise (Réfs. 1 à 6). Malgré le fait que les objectifs de cette nouvelle technique demandent la réduction des vibrations, l'élimination du décrochage, l'absorption des interférences aérodynamiques, la suppression des perturbations atmosphériques et l'introduction de la stabilité artificielle (Fig. 1), l'impact sur les qualités de vol risque d'être considérable et ouvre des possibilités nouvelles aux améliorations des qualités de pilotage des hélicoptères actuels. Nous assisterons dans l'avenir, non seulement à l'extension du domaine de vol, mais aussi à l'allègement de la tâche du pilote par l'introduction des boucles internes, travaillant indépendamment pour améliorer les réponses dynamiques de l'hélicoptère et pour éviter les effets nuisibles des couplages entre les degrés de liberté. En ayant cette évolution des commandes en mémoire, rappelons que l'hélicoptère moderne, en comparaison avec ceux volant il y a dix ans, présente des tendances de définition nouvelles et certains changements de la doctrine opérationnelle rendant les problèmes de qualités de vol nettement plus sévères. Nous pouvons distinguer cinq tendances :

- La charge sur le disque a presque doublé au cours des derniers dix à quinze ans, entraînant, en conséquence, l'augmentation du niveau énergétique du sillage et provoquant, à son tour, des distorsions plus élevées des charges aérodynamiques et une dissipation considérable de puissance (Réfs. 7 et 8).

- La mission du vol en rase-mottes (NOE) a été adoptée pour assurer une relative sécurité grâce à l'effet de masque des accidents de terrain et est devenue particulièrement exigeante en ce qui concerne les qualités de vol des hélicoptères militaires modernes.

- En réponse à la demande de missions de plus en plus difficiles, l'industrie propose des ADAV d'avant-garde, caractérisés par l'adjonction des ailes, par le basculement des rotors, par l'addition des moyens de propulsion auxiliaires (ABC), par l'arrêt du ou des rotors, (X-Wing, Tilt Rotor), par l'arrêt et l'escamotage du rotor (Réf. 9). Ces concepts nouveaux, structuralement et fonctionnellement plus complexes, introduisent, en outre, la transition et nécessitent un effort important d'allègement des tâches du pilote.

- Corrélativement, l'expansion du domaine de vol de l'hélicoptère classique est devenue une exigence prioritaire. Pour atteindre des paramètres d'avancement (μ) et des coefficients de portance (Cz_{moy}) les plus élevés possibles, l'hélicoptère actuel fonctionne et manoeuvre aux frontières et souvent au-delà des frontières d'accessibilité (Réf. 13). La CAG permet dans de tels cas d'assouplir les contraintes existantes et de rendre la manoeuvre possible dans des situations autrement interdites.

En conséquence, une tendance générale est née de dépasser les limitations classiques de l'hélicoptère pour élargir ainsi son domaine de vol (Réfs. 1, 2 et 4).

- L'avantage principal de la CAG, introduit sur les voilures tournantes, permet de procéder à un degré supérieur de l'optimisation des conditions de fonctionnement de l'hélicoptère. Cette optimisation est obtenue par la détection des conditions de fonctionnement et de charge, par le traitement adéquat de cette information et, finalement, par l'utilisation du signal résultant dans la boucle de retour de la commande classique ou nouvelle en vue de l'amélioration du comportement dynamique de l'appareil, Fig.2. La référence 5 donne un certain nombre d'exemples spécifiques de l'application de cette technique.

(*) FOR ENGLISH VERSION SEE PAGE 4.

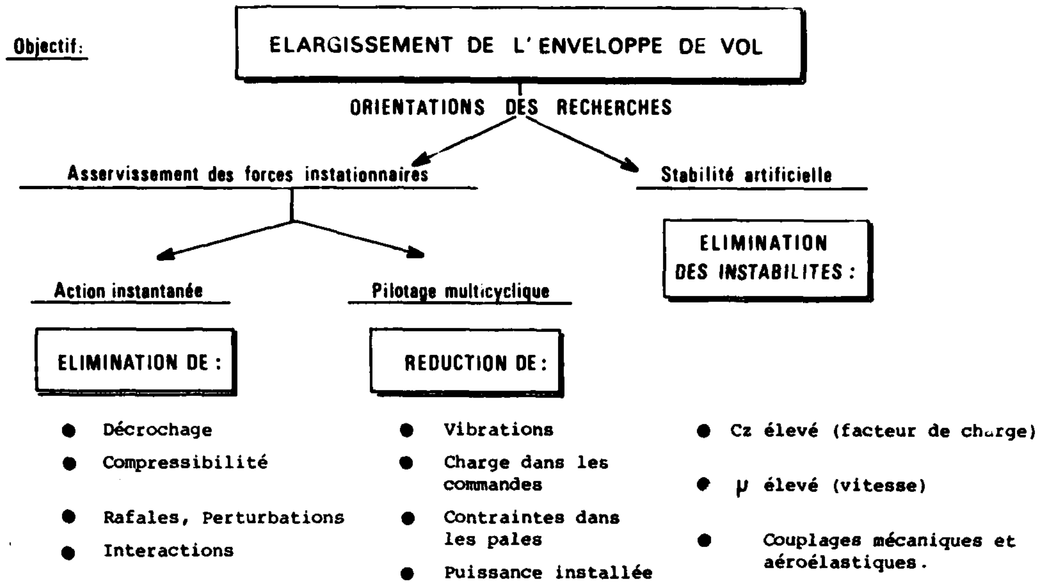


Fig. 1 - La CAG des rotors d'hélicoptère

Remarquons ici que, dans la recherche des performances supérieures de la manoeuvrabilité et de la maniabilité de l'hélicoptère, la théorie de la commande optimale, utilisant les filtres de Kalman, montre une grande utilité (Réf. 14). Cette approche est rendue d'autant plus facile que la découverte de la matrice de transfert du rotor a fourni un modèle analytique particulièrement simple (Réf. 15).

Dans cette communication, nous nous concentrerons pour étudier l'étendue de l'impact de la CAG sur les qualités de vol des hélicoptères et mentionnerons des critères qui permettront la qualification des solutions nouvelles envisagées.

2. - LES LIMITATIONS DES COMMANDES ACTUELLES DE L'HELICOPTERE

La commande classique de l'hélicoptère utilise le plateau cyclique créant une variation monocyclique du pas de pales. La position du plateau cyclique peut être modifiée librement par le pilote ou par le pilote automatique. Ce système introduit une double limitation :

- d'une part, les variations du pas sont pratiquement limitées à des variations une fois par tour du rotor.
- d'autre part, les variations de pas de toutes les pales sont périodiquement les mêmes à cause du couplage mécanique provenant du plateau cyclique.

Les références 1, 2 et 4 montrent comment ces limitations peuvent être éliminées. Il sera donc possible dans l'avenir d'introduire le pilotage multicyclique pour réduire les vibrations et utiliser un concept nouveau, commande individuelle de la pale, pour éviter le décrochage et créer la stabilité artificielle.

La CAG s'oriente, en plus, vers des solutions encore plus révolutionnaires comprenant l'élimination du plateau cyclique et l'installation des vérins électrohydrauliques dans la partie tournante de la tête du rotor.

Toutefois, dans l'immédiat, la CAG comprend le plateau cyclique et la cinématique classique. Le pilotage multicyclique et la commande individuelle de la pale s'introduiront grâce aux oscillations imposées à la partie non-tournante du plateau cyclique.

3. - APPLICATION DE LA CAG AUX VOILURES TOURNANTES

Les dispositifs de CAG projetés au cours des années 70 représentent des systèmes comprenant les pilotes automatiques et des augmentateurs de stabilité d'origine plus ancienne. Les nouveaux dispositifs ont un champ d'application nettement plus large et sont caractérisés par une bande passante supérieure à 30 Hz. Le traitement de l'information captée est plus élaboré, plus sophistiqué, agit à l'insu du pilote pour remplir des tâches hors des aptitudes du pilote ou des tâches soulageant son travail.

A la base, la technologie de la CAG, telle qu'elle est envisagée sur les hélicoptères, ne diffère pas de celle appliquée aux voilures fixes : la théorie, les approches analytiques, la technique des calculateurs numériques et des vérins à large bande passante, les commandes électriques ou par fibres optiques, sont analogues à celles des avions. La différence caractéristique provient du fait que les phénomènes intéressant les rotors sont périodiques et peuvent être décrits par un système

d'équations différentielles linéaires dont les coefficients sont, toutefois, périodiques (variables avec le temps). Ce fait introduit un certain degré de souplesse dans le traitement analytique de la dynamique du rotor avec la possibilité de procéder par itérations, tour par tour, pour atteindre progressivement un état optimisé satisfaisant aux critères de performance prédéterminés. Plus particulièrement, la découverte de la matrice de transfert du rotor fournit une relation très simple entre les coefficients de Fourier caractérisant les signaux d'entrée et de sortie. Cette matrice de transfert constitue un modèle analytique très pratique dans les applications de la théorie de la commande optimale stochastique, Réfs 14 et 15. Ainsi, des moyens analytiques très puissants sont disponibles dans la recherche des qualités de vol supérieures, au moment où la demande s'y fait sentir de plus en plus pressante.

4. - ASSERVISSEMENT ANTI-DECROCHAGE, UN EXEMPLE D'APPLICATION DE LA CAG

L'application de la CAG aux rotors d'hélicoptères, Fig.1, montre un caractère multidirectionnel, très varié par ses objectifs. L'impact sur les qualités de vol se ressentira donc sur les niveaux différents avec un degré d'influence différent. Les dispositifs signalés sont actuellement dans leur phase de recherche et une revue exhaustive semble encore prématurée. Nous allons donc prendre un exemple dont l'influence sur les qualités de vol nous semble la plus importante et nous analyserons un asservissement basé sur la commande individuelle de la pale pour éviter le décrochage de la pale reculante, asservissement anti-décrochage. Nous prendrons l'élimination du décrochage comme cas typique d'application de la CAG et montrerons l'impact que ce concept peut avoir sur les qualités de vol de l'hélicoptère. Ce concept, actuellement expérimenté en soufflerie, Réf. 6, est capable d'améliorer considérablement les qualités de manoeuvre grâce à l'économie de puissance qu'il peut apporter. Son idée de base est simple et consiste en la réduction du pas de la pale lorsque le coefficient de portance local C_z atteint des valeurs avoisinant le décrochage. La figure 3 montre un cas bi-dimensionnel, qui peut être facilement étendu au cas tri-dimensionnel. Sur la figure 3 nous remarquons que pour le même Cz = 1,1, la traînée varie dans le rapport 1 à 10. Aussi, pour des raisons de simplicité de la démonstration, nous admettrons des variations quasi-statiques de la portance et de la traînée. Considérons deux cas de vol identiques, différant uniquement par la manière d'obtention de la portance. Le premier cas est classique, le second réalise la même variation de la portance sans décrochage, grâce à un contrôle adéquat de l'angle d'attaque maintenant le point de fonctionnement sur la partie linéaire de la caractéristique de Cz. Il devient alors évident que nous réaliserons une économie de puissance dans le second cas. Une telle économie de puissance a été démontrée expérimentalement au cours des essais récents en soufflerie. Dans ce cas typique, le gain énergétique atteint 8% pour μ = 0,3 et Cz_{moy} = 0,6, Fig. 4. L'étude théorique d'un cas analogue : μ = 0,3, Cz_{moy} = 0,615, a montré un gain de puissance de 10% accompagné d'un gain possible de portance de 5%, Fig. 5. Aux grandes vitesses, les gains de puissance et de portance permettront d'augmenter la marge entre la puissance en palier et la puissance maximale disponible avec effet bénéfique pour l'exécution des manoeuvres lentes, à facteur de charge élevé. La figure 6, reproduite de la Réf. 13, montre la variation de l'angle d'attaque local, à 80% du rayon de la pale. Même dans les cas aussi extrêmes, le décrochage peut être évité grâce à la commande active et apporter une marge énergétique nettement supérieure. Les limitations dues au décrochage peuvent être surmontées en apportant l'augmentation de la manoeuvrabilité de l'appareil. Plus particulièrement, la zone des "pertes dues au décrochage", qui figure sur la figure 7, correspond au point, où les effets dûs au décrochage deviennent significatifs. Les conditions de vol dans cette zone montrent un accroissement rapide de la puissance demandée et, sur certains types de rotor, des contraintes élevées dans les pales, le flottement des pales, des problèmes de perte de contrôle. Cette zone définit par ailleurs, le point de référence limite en Cz_{moy} pour des manoeuvres lentes au-delà duquel les besoins en puissance installée deviennent excessifs, Réf. 13. Il est possible de déplacer cette zone vers la limite extrême "limite aérodynamique de Cz_{moy}", liée au Cz_{max} du profil de pale, avec l'avantage de l'augmentation de la marge de puissance disponible et de l'élimination du flottement des pales.

D'après la figure 1, l'élimination du décrochage représente une des multiples possibilités de la CAG, capables d'apporter des avantages substantiels en améliorant les performances dynamiques de l'hélicoptère. Parmi les diverses boucles d'asservissement signalées, l'asservissement anti-décrochage exercera une influence prépondérante sur les qualités de vol grâce à l'apport d'une marge énergétique nettement plus élevée.

Nous examinerons ensuite dans quelle mesure l'introduction de la CAG peut modifier les critères de pilotage.

5. - L'IMPACT DE LA CAG SUR LES QUALITES DE VOL

L'introduction de la CAG ne semble ni apporter, ni nécessiter des modifications des critères déjà existants ou des approches utilisées pour leur établissement. Toutefois, nous pensons que la théorie moderne de la commande optimale stochastique utilisant des filtres de Kalman ou des algorithmes équivalents rendra d'une part, l'analyse des qualités de vol plus approfondie et, d'autre part, plus objective.

L'optimisation utilise une fonction de coût qui fournit la base pour notre jugement. Les matrices de pondération constituent la partie arbitraire, subjective des contraintes, fruit d'une expérience passée. Une fois définies, les matrices nous donnent un paramètre de qualité instantanée, caractérisant simultanément l'appareil, la commande et le pilote. Dans ce cas, les critères deviennent un seul critère, résultat pondéré de tous les critères dont nous disposons quantitativement.

La performance ne forme pas, toutefois, le seul facteur qui qualifie l'appareil. Il est indispensable, en outre, de prendre en compte l'effort fourni par le pilote pour accomplir cette performance. Nous somme amenés ainsi à revenir à la procédure de quotation du pilote ou à établir un modèle analytique du pilote humain tenant compte des effets de fatigue et de la dégradation de la performance du pilote en fonction aussi bien de la durée que de la difficulté de sa tâche.

Si aucune solution globale ne pointe encore à l'horizon, nous pouvons raisonnablement nous attendre à l'amélioration de la compréhension des phénomènes fondamentaux et du traitement analytique des problèmes complexes de l'opérateur humain, amélioration que l'application de la CAG apporterait au cours de la prochaine décade.

6. - CONCLUSION

En conclusion, les limitations de l'hélicoptère actuel seront dépassées dans l'avenir grâce aux systèmes nouveaux, orientés vers les commandes automatiques généralisées, caractérisées par des asservissements à boucles multiples, optimisées et auto-adaptatives, qui permettront, à leur tour, une optimisation à un niveau plus élevé, des qualités de vol. Parmi les multiples asservissements envisagés, l'asservissement anti-décrochage constitue le système dont l'impact sera le plus prononcé sur les qualités de vol. Ses effets bénéfiques s'exerceront sur la marge de puissance grâce à un excès énergétique que l'élimination du décrochage peut apporter.

La recherche concernant la CAG est caractérisée par un vaste champ d'application, mais demande un effort considérable de développement pour s'intégrer aux hélicoptères opérationnels.

7. - REFERENCES (Voir page 7)

THE IMPACT OF ACTIVE CONTROL ON HELICOPTER HANDLING QUALITIES

Marcel KRETZ

Chief Engineer

Giravions Dorand Industries

92150 Suresnes, France

ABSTRACT

Notable changes have occurred during the last decade in the concepts envisaged for controlling the working conditions of helicopter rotors. New trends, prompted by the techniques of active control applied to fixed wings, are oriented towards the automatic control of dynamic phenomena (vibration, instability) and aerodynamic phenomena (stall effects, interaction, gusting). These new trends feature frequency responses much wider than those of conventional autopilots, extending up to 30 Hz. At the same time, contemporary military helicopter design has become much more demanding with regard to handling qualities : higher disk loading, NOE mission requirement, advent of advanced rotor aircraft concepts (tilt rotor, ABC, X-wing, Stoppable and stowable rotors) and a general broadening of the flight envelope (higher μ and higher C_L).

The paper analyzes the impact of active control on handling qualities and discusses the benefits that may be reaped from their implementation. In conclusion, present-day control system limitations due to the use of monocyclic swashplate principles will have to be removed in the future by unconventional control systems based on multiloop self-adaptive control resulting in higher-order optimization of handling qualities.

1. - INTRODUCTION

The last decade has witnessed the preparation of major developments in helicopter rotor control. Prompted by the techniques of active control applied to fixed wings, designers have attempted to widen frequency responses to encompass control dynamic and aerodynamic phenomena far beyond the monocyclic capabilities of the conventional helicopter rotor swashplate (Refs. 1 through 6). Although the main goals of these new techniques include reduced vibration, the elimination of stall conditions, the absorption of aerodynamic interference, the elimination of external disturbance and the implementation of artificial stability (Fig. 1), their impact on handling quality will be considerable, providing new opportunity to uprate the flying qualities of present helicopters. In the future, we shall not only see extended flight envelopes but also simplification of the pilot's task by the introduction of inner independently operating loops improving the dynamic response of the helicopter and eliminating the prejudicial effects of response coupling. While considering this development of flight control techniques, it should be borne in mind that today's operational doctrines compared with those of the past raise much more severe handling problems. Five distinct areas of development tendencies may be identified :

- Disk loading has nearly doubled over the last ten years, with a subsequent increase in rotor-wake power levels producing in turn higher air load and power dissipation distortion (Refs. 7 and 8).

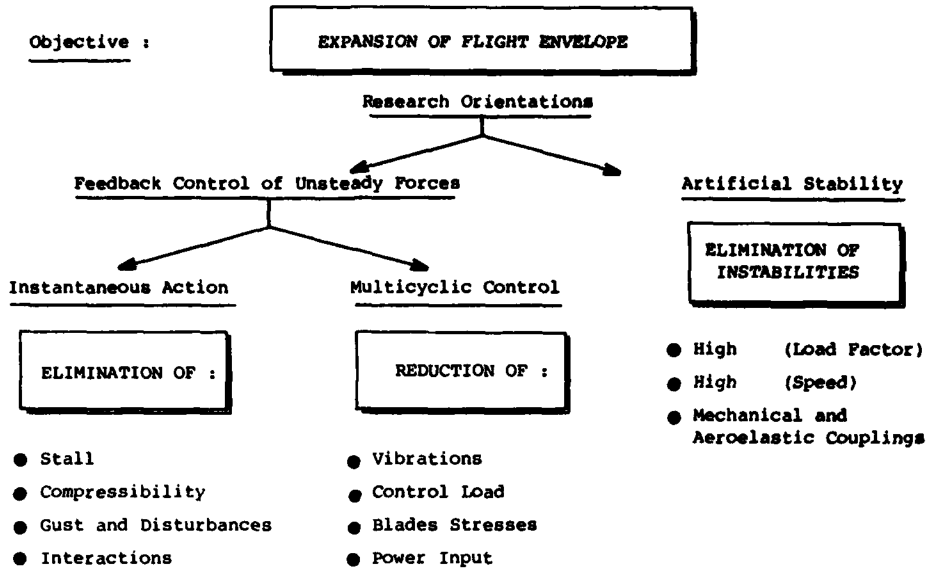


Figure 1 - ACTIVE CONTROL OF HELICOPTER ROTORS

- The nap-of-the-earth (NOE) mission, adopted to take advantage of the relative protection afforded by terrain masking, is highly demanding in maneuverability of present-day military helicopters (Refs. 10, 11 and 12).

- In order to meet these ever-increasing operational requirements, advanced VTOL rotor concepts appeared : the addition of fixed wings, tilting rotors, addition of auxiliary propulsion (ABC), rotor stopping in flight (the X-wing and tilt-rotor), rotor stopping and stowage in flight (Ref. 9). Moreover, these concepts requiring greater structural and functional complexity involve transition flight conditions making it essential to alleviate the pilot's task.

- At the same time, extension of the pure helicopter flight envelope constitutes a high-priority requirement. Striving towards higher advance ratios (μ) and higher blade loading (C_T), the modern helicopter is operating and maneuvering on and sometimes beyond the boundaries of accessibility areas (Ref. 13). Here again, active control can relieve natural constraints, rendering a particular maneuver acceptable in otherwise unacceptable situations. Consequently, there is a general tendency to relax present-day helicopter limitations and broaden the flight envelope (Refs. 1, 2 and 4).

- The prime advantage of active control as applied to rotary wings is the higher-order optimization of helicopter working conditions. Such optimization is achieved essentially by sensing the helicopter working and loading status, processing the resulting information in an appropriate manner and then feeding it back to a conventional or unconventional control system for improving the dynamic behavior of the craft (Fig. 2). Ref. 5 gives specific examples of the application of these techniques to rotary wings. It should be mentioned here that Kalman's optimal estimation theory is extremely valuable in the quest for the improved handling qualities of helicopters (Ref. 14). This was made possible by the development of an extremely simple analytical model of the rotor : the rotor transfer matrix (Ref. 15).

This paper concentrates more particularly on the extent of the impact that feedback may have on the handling qualities of future helicopters and mentions additional criteria that could help in the evaluation of new technical propositions.

2. - LIMITATIONS OF PRESENT HELICOPTER ROTOR CONTROL

Conventional helicopter rotor control is achieved by the use of a swashplate system producing monocyclic blade pitch variation. The setting of the swashplate can be modified as required by the pilot or an autopilot. Such a system, however, has two major limitations :

- Firstly, pitch variations are practically limited to one per revolution.

- Secondly, the periodical variations of all blades are the same, since they are produced by the same swashplate. References 1, 2 and 4 show how such limitations can be overcome and how in the future it will be possible to introduce multicyclic pitch variation to reduce vibration and make use of a new concept : Individual Blade Control (IBC) to avoid stall and create artificially enhanced stability. In addition, active control offers the promise of still further conceptual changes of a revolutionary nature, including elimination of the swashplate and the use of electrohydraulic actuators in the

rotating part of the rotor hub. In the near future, however, the active control or rotary wings will retain the swashplate and conventional linkages. The IBC inputs of higher harmonic content will be fed to the rotor by appropriately controlled oscillation of the non-rotating part of the swashplate.

3. - APPLICATION OF ACTIVE CONTROL TO ROTARY WINGS

The type of active control proposed in the seventies involved systems comprising autopilots and stability-augmentation devices developed earlier. Today's proposals have a much wider range of application and possess a much more extended frequency band of at least 30 Hz. The processing of signals is much more elaborate and sophisticated, performed independently of pilot action, for executing tasks beyond human scope or assisting the pilot to assume his workload. Basically, active control techniques as applied to rotary wings are closely similar to those applied to fixed wings with regard to theory, analytical approach, the use of microprocessors, wide-band actuators and fly-by-wire or fly-by-light controls. The main differences arise from the fact that rotor phenomena are periodic and can be described by a system of linear differential equations with periodic (time-variable) coefficients. This fact introduces a certain degree of flexibility in the analytical treatment of rotor dynamics with the possibility of applying iterative methods to achieve an optimized state satisfying predetermined performance criteria by successive approximation. In particular, the discovery of the rotor transfer matrix provided an easily manageable relationship between the input and output Fourier coefficients. This rotor transfer matrix constitutes an analytical model highly suitable in applications of the Kalman's optimal estimation theory (Refs. 14 and 15). Powerful tools are thus available to assist in the search of improved flying quality in the face of the severe specifications generated by new requirements.

4. - STALL BARRIER FEEDBACK : AN EXAMPLE OF ACTIVE CONTROL APPLICATION

The application of active control to helicopter rotors (Fig. 1) is of a multipurpose nature with many facets and orientations. The impact on handling qualities is felt at different levels and to different degrees of effectiveness. Systems being presently studied are at the research stage and it would appear too early to review all of them. We shall therefore consider an example that could have considerable influence on handling qualities and concentrate on an IBC-based feedback system avoiding stall on the retreating side of the rotor disk. Stall-avoidance is taken as a typical application of active control to helicopter rotors and illustrates the impact that this concept could have on helicopter handling qualities. According to experimental data produced by windtunnel investigations (Ref. 6), this concept has considerable potential for improving the maneuvering ability of a helicopter by appreciable power savings. Known as Stall Barrier Feedback or simply SBF (Ref. 6), this concept is based on the elementary idea of reducing blade pitch when the local C_L reaches values close to those of stall onset. Fig. 3 shows a 2-D case which may easily be extended to the 3-D case. It is seen that the drag is ten times more for the same C_L of 1.1. For the sake of simplicity, we shall consider the quasi-steady-state lift and drag variations. We shall consider two identical flight configurations, differing only by the manner of creating lift. The first is conventional, whilst the second achieves the same lift variation without stalling by appropriate blade pitch control on the linear part of the C_L characteristic. It is obvious that the second saves power, as has been demonstrated by recent windtunnel testing. Typically, such saving can amount to 8% with an advance ratio of 0.3 and a lift coefficient of 0.6 (Fig. 4). Theoretical analysis of a similar flight configuration with $\mu = 0.3$ and $C_L = 0.615$ resulted in a power saving of 10% with a 5% increase in lift (Fig. 5). At high forward speeds, the power savings and lift gains will increase the margin between level-flight power and maximum available power, with beneficial effects during sustained maneuvers. Fig. 6 reproduced from Ref. 13 shows the azimuthal distribution of blade angle-of-attack at 80% radius. In this extreme case, deep stall can be avoided by the use of active control techniques and provide a much wider power margin. The limits due to stall boundaries can thus be pushed back with appreciably improved maneuverability. In particular, the stall-losses region (Fig. 7) shows the point at which the effects of retreating blade stall become significant. Flight behavior in this region is characterized by a sudden increase in required power and, in the case of certain types of rotor, high blade loading, stall, flutter and controllability problems. It also indicates the maximum design point for sustained maneuvers because of the excessive installed power requirements at high C_L (Ref. 13). This stall region can be moved towards the ultimate boundary, the aerodynamic C_L limit defined by the maximum C_L of the airfoil, thus widening the excess power margin and eliminating the danger of stall flutter.

As shown in Fig. 1, stall elimination is achieved by one of several feedback characteristics capable of significantly improving the dynamic performance of helicopters. Of these different feedback characteristics, however, SBF will probably have the greatest impact on handling qualities by the substantial increase in the available power margin.

We shall now examine the extent to which the introduction of active control can affect handling-quality criteria.

5. - THE IMPACT OF ACTIVE CONTROL ON HANDLING-QUALITY CRITERIA

It would appear that the introduction of active control does not require basic modification of already existing criteria or of analytical approaches used for establishing flying-quality criteria. There is hope, however, that Kalman's optimal control theory might rationalize the analysis to a greater extent and render the results more objective.

The optimization technique involves a cost function that provides the basis of our judgement. The weighting matrices constitute the arbitrary, the subjective part of constraints, the result of previous experience. Once these matrices have been defined, we possess an instantaneous qualification parameter that simultaneously defines the aircraft, the control system and the pilot. In this case, the criteria reduce to a single analytical criterion, a weighted result of all available criteria in a quantitative form. Performance, however, is not the only factor qualifying an aircraft. We must also consider the pilot workload required to achieve a given level of performance. So we are driven back to pilot quotation procedure or to establishing a human pilot model that includes fatigue effects and degradation of the pilot's dynamic performance with time and in the face of difficulty. Although no complete solution to the problem of establishing handling-quality criteria can be expected from the application of active control,

improved understanding of the basic phenomena as well as of the analytical treatment of complex human-operator problems is foreseeable in the next decade.

6. - CONCLUDING REMARKS

It may be stated in conclusion that present-day helicopter limitations resulting from the mono-cyclic swashplate must be overcome in the future by unconventional systems oriented towards multiloop self-adaptive controls leading in turn to the higher-order optimization of handling qualities. At the same time, the introduction of active control will contribute to better understanding and more accurate definitions of criteria ascribing pilot tasks and workload. Our present research has a wide range of applications but will require considerable development effort to become operational.

7. - BIBLIOGRAPHICAL REFERENCES

1. Kretz M. : "Relaxation of Rotor Limitations by Feedback Control", 33rd Forum of the AHS, May 1977.

2. McHugh F.J. and Shaw J. : "Helicopter Vibration Reduction with Higher Harmonic Blade Pitch" 3rd European Rotorcraft Forum, Sept. 1977.

3. Wood E.R., Powers R.W. and Hammond C.E. : "On Methods for Application of Harmonic Control", 4th European Rotorcraft Forum, Sept. 1978.

4. McCloud III J.L. and Weisbrich A.L. : "Wind Tunnel Test Results of a Full-Scale Multicyclic Controllable Twist Rotor", 34th Forum of the AHS, May 1978.

5. Kretz M. and Larché M. : "Future of Helicopter Rotor Control", 5th European Rotorcraft Forum, Sept. 1979.

6. Kretz M. : "Active Elimination of Stall Conditions", 37th Forum of the AHS, May 1981.

7. Sheridan P.F. and Smith R.P. : "Interaction Aerodynamics : a New Challenge to Helicopter Technology", Journal of the AHS, Jan. 1980.

8. Roesch P. and Vuillet A. : "New Designs for Improved Aerodynamic Stability on Recent Aérospatiale Helicopters", 37th Forum of the AHS, May 1981.

9. Kretz M. : "Stoppable and Stowable Jet-Flap Rotor Concept", 4th European Rotorcraft Forum, Sept. 1978.

10. Dooley L.W. : "Handling Qualities Considerations for NOE Flight", Journal of the AHS, Oct. 1977.

11. Chen R.T.N. : "A Piloted Simulator Investigation of Augmentation Systems to Improve Helicopter NOE Handling Qualities", 34th Forum of the AHS, May 1978.

12. Shupe N.K. : "The Night Navigation and Pilotage System", 37th Forum of the AHS, May 1981.

13. Wells C.D. and Wood T.L. : "Maneuverability : Theory and Application", Journal of the AHS, Jan. 1973.

14. Taylor R.B. and Farrar F.A. : "An Active Control System for Helicopter Vibration Reduction by Higher Harmonic Pitch", 36th Forum of the AHS, May 1980.

15. McCloud III J.L. and Kretz M. : "Multicyclic Jet-Flap Control for Alleviation of Helicopter Blade Stresses and Fuselage Vibrations", Rotorcraft Dynamics, NASA SP-352, Feb. 1974

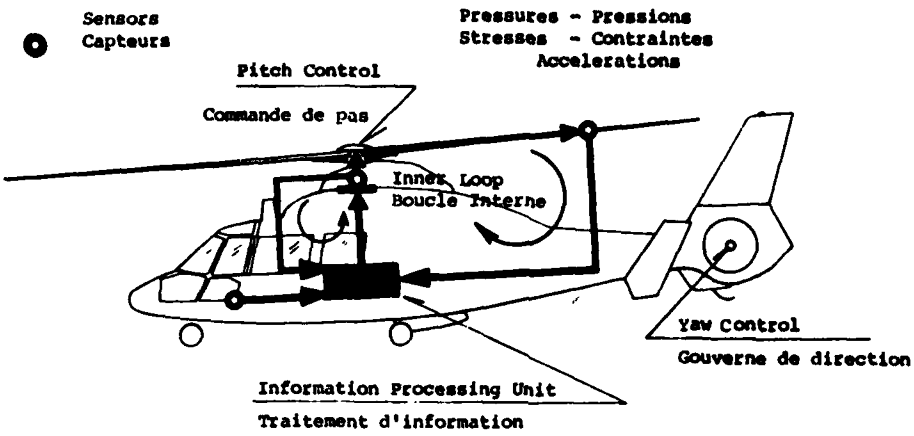


Fig. 2 - Principles of Active Control
Principes de la CAG

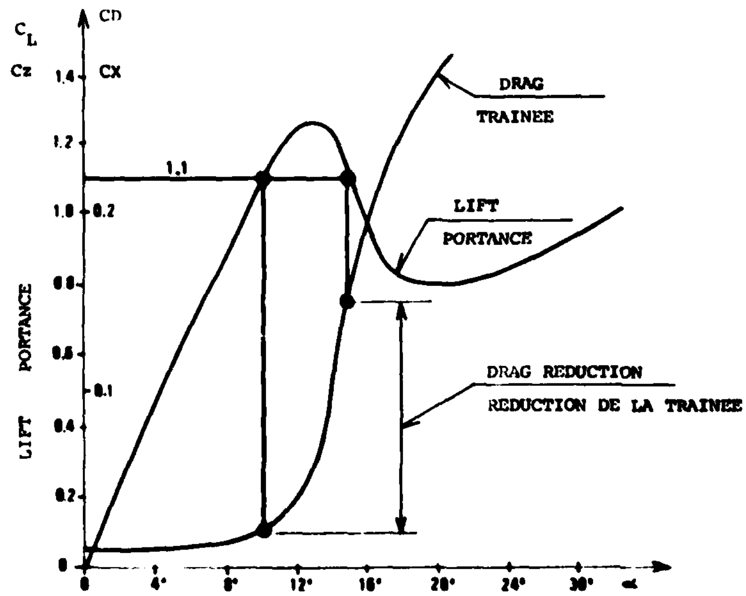


Fig. 3 - NACA 0012 LIFT-DRAG CHARACTERISTICS, M = 0.3
Polaire du profil NACA 0012

Doc. C-23

Fig. 4 - Stall Barrier Feedback.Test Recordings
Asservissement Anti-décrochage.Enregistrements d'essais

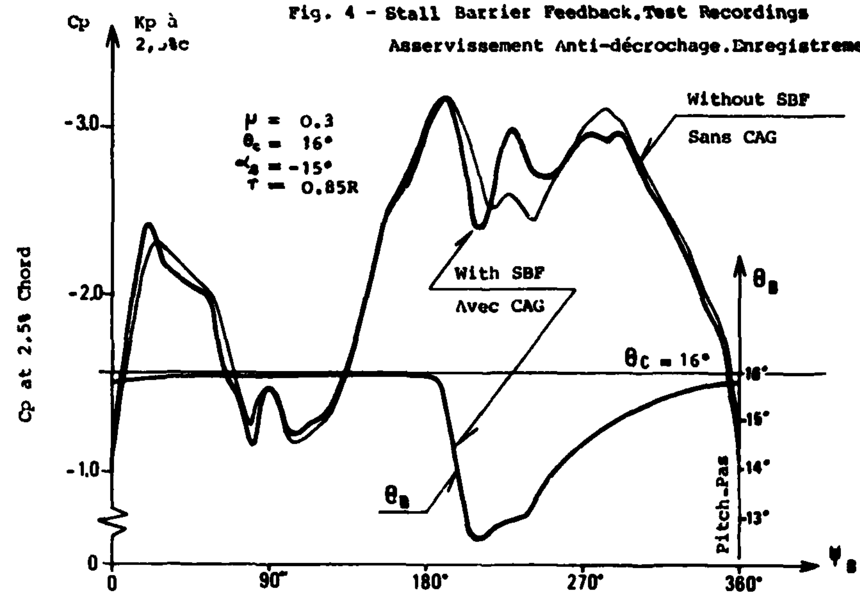
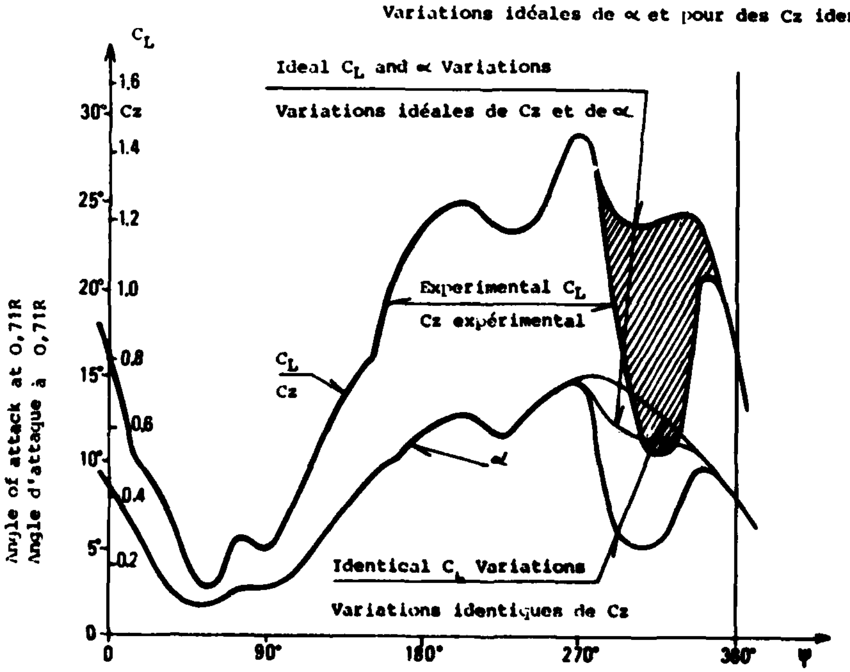


Fig. 5 - Ideal and Identical C_L Angle of Attack Variations
Variations idéales de α et pour des Cz identiques



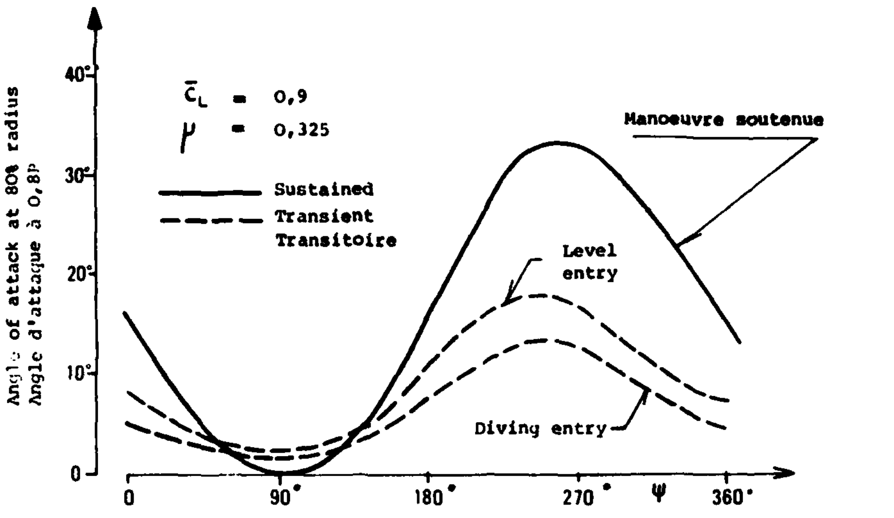


Figure 6 - Angle of attack comparison, sustained and transient maneuvers, level and diving entry
Comparaison entre les angles d'attaque pour des manoeuvres transitoires et soutenues.

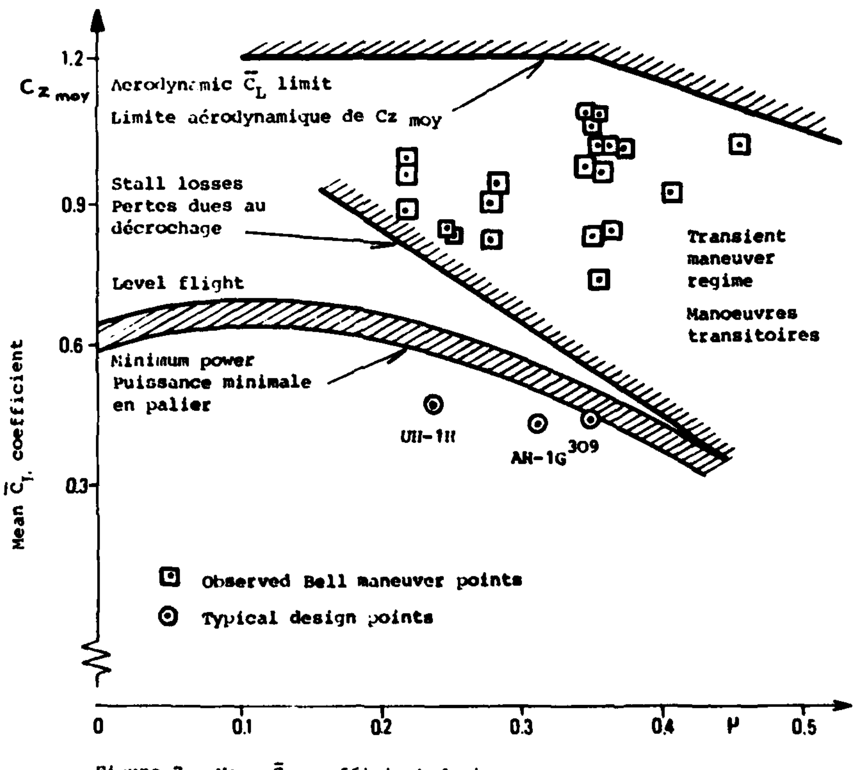


Figure 7 - Mean $\bar{C}_L$ coefficient during maneuvres
Cz_{moy} pendant manoeuvres

OPERATIONAL CRITERIA FOR THE HANDLING QUALITIES OF COMBAT HELICOPTERS

by

Major W Steward AAC

HQ Director Army Air Corps

Middle Wallop

Stockbridge

Hampshire SO20 8DY, UK

SUMMARY

To minimize the threat from air and ground based weapon systems, combat helicopter operations require the use of concealed low level flight. The paper discusses the tasks facing the combat helicopter pilot during a typical anti-armour mission and concludes that, by reducing the flying workload, assisting in the exploitation of maximum aircraft performance, and enhancing control accuracy, better handling qualities can contribute to improved operational effectiveness. Primary consideration is given to daylight operations in VMC, but the requirements for missions at night and in adverse weather, and for training are also addressed, together with the implications for handling qualities posed by the threat of armed helicopters in the air to air role.

1. INTRODUCTION

The British Army Air Corps' current battlefield helicopters, the Lynx AH MK 1 and Gazelle AH MK 1, entered service in the 1970's to fulfil a variety of roles. Changing priorities have resulted in the Lynx becoming primarily an anti-armour helicopter, fitted with a roof mounted sight and the TOW missile system; the Gazelle is soon to be fitted with an observation sight to enable it to perform its target acquisition task as part of the anti armour team.

The change of Lynx to the anti armour role, with its emphasis on very low level flight and sustained hovering has caused a review of a number of aspects, including handling qualities upon which this paper is based. The views stated are, however, solely those of the author.

Continuing development in the variety, numbers, and sophistication of weapon systems to counter the successful employment of helicopters on the European battlefield indicates that low level flight at and below obstacle height, to aid concealment, achieving surprise and reducing vulnerability, is likely to remain the primary combat helicopter tactic. Payload restrictions will limit the amount of hardening which can be given to the helicopter; to survive, the helicopter will need to avoid being hit rather than be capable of tolerating considerable damage. Although improved stand off capability, countermeasures, and warning systems will reduce vulnerability, it is agility derived from the inter-related performance and handling qualities which gives the combat helicopter its inherent battleworthiness.

The combat helicopter will continue to have competition for funding from other land and air systems; improvements to handling qualities must therefore be realistic, and affordable. Although Active Control Technology holds considerable promise, it may be necessary to accept a degree of pilot compensation. Similarly, achieving the right balance between integrity, cost, and complexity may result in reversionary modes with added pilot workload.

2. THE COMBAT MISSION

Depending finally upon the distances involved, an anti-armour helicopter mission could last for between 1 and 3 hours. For the purpose of considering handling qualities, this may be grouped into three main phases, each of similar duration; fast low level flight, slower nap-of-the-earth (NOE) flight, and the sustained hover in the fire or observation position. There are also periods of ground running, at the start point and possibly at a final briefing point short of the fire or observation position. Although the reconnaissance helicopter could be expected to spend longer in the hover, the major difference from the handling qualities point of view will be the increased emphasis on the need for pilot workload reduction during the hover. During all the phases of the sortie, the emphasis will be on both 'head-up' operation of the aircraft and the correct trade-off between tactical reaction time and concealment.

3. TAKE OFF, LANDING, AND GROUND MANOEUVRES

Enhanced ground mobility is desirable for the benefits it brings to survivability, through easier concealment, as well as ground movement during in-barracks training and maintenance. The combat helicopter will thus probably have a wheeled undercarriage, although it is doubtful whether wheels of an acceptable size will confer more than limited mobility other than on reasonably level firm ground.

The combat helicopter must be stable and free from resonance during ground taxying, take off, and landing on all likely operational surfaces, including slopes in the order of 10°, and in all wind directions. The rotor must be able to be engaged and stopped under the same conditions, and to facilitate the use of maps and documents, as well as the completion of other cockpit activities, hands-off ground running with rotors turning should be possible.

The need for concealment may result in the mission starting from an obstructed landing site such as woodland or built-up areas thus defining the required out-of-ground-effect performance. Maximum operational flexibility will require full fuel and armament loads, resulting in initial take-offs at maximum all up weight. When the associated loss of hover capability, speed, and agility are acceptable, a battle overload may be used to temporarily increase payload. Performance considerations notwithstanding, the combat helicopter must have safe predictable handling qualities during take off and landing. The directional control of the helicopter must permit take off and landing in all relative wind directions so that the flight path may conform to the ground, or tactical situation.

4. LOW LEVEL AND CONTOUR FLIGHT

The first phase of the move forward to the contact zone, and last on the return, will probably be flown with the emphasis on speed rather than on absolute concealment. Terrain will dictate the actual mode of flight which will probably involve a mixture of low level flight, at sensibly constant speed and a steady height clear of obstacles, and contour flight again at nominally constant speed but with height varying to follow the major terrain features. The crew will still be settling into the sortie, and a hands-off cruise capability, involving height and heading holds, would permit maximum attention to be given to continued planning and preparation, as well as look-out.

When the tactical situation requires a minimum reaction time, the maximum dash capability of the helicopter will be used, care being taken not to prejudice the mission through loss of concealment.

High ground speeds over undulating and broken terrain will require the use of low 'g' and occasional negative 'g' to allow the pilot to follow contours whilst minimizing exposure. Flight trials with the Lynx have shown that negative 'g' values of - 0.5 can be held for several seconds during push-over manoeuvres, with transient values in the order of - 1.0 'g' being reached. Negative 'g' flight was found to be of benefit to terrain following at air speeds as low as 60 knots, and the combat helicopter must have handling qualities which permit the safe and easy use of this flight regime.

Despite thorough pre-flight preparation to supplement knowledge of the ground, and careful look-out by the crew, unexpected obstacles such as unmarked wires will require the crew to take rapid avoiding action, often involving the use of high load factors. Such accelerations frequently cause the rotor to accelerate into an autorotative condition, requiring pilot intervention to prevent an overspeed. In addition, at low level, flight path control becomes very demanding at large bank angles where the control of height excursions requires considerable pilot activity. These are obviously undesirable characteristics, and a means of controlling rotor speed transients and minimizing height excursions would confer more relaxed handling qualities, and perhaps facilitate the use of enhanced manoeuvrability.

Ideally, the pilot should not encounter an aircraft limit during normal manoeuvres. Combined with good engine response, and an absence of rotor speed control problems, this would result in a carefree manoeuvre capability, freeing pilot attention for operational tasks. Where the intrusion of a manoeuvre limit in unavoidable, the required attention to cockpit instruments should be minimized, implying the use of audio warnings or helmet mounted displays. Flight control systems, including suitable control force characteristics, and possibly using active control technology, should be able to assist with the exploitation of the full flight envelope. However, unlike the use of envelope limiting control systems for upper air work in high performance fixed wing aircraft, the combat helicopter pilot needs to be able to exceed an envelope limitation to save the aircraft rather than fly into an obstacle within the limit. The problem should be avoidable if the combat helicopter is capable of manoeuvring at 3 to 4 'g' without reaching a limitation; this may require the use of lift compounding.

5. NOE FLIGHT

The next phase of the mission may be considered to apply when the consequences of exposure become more critical, and the need for concealment is paramount. Although speed is sacrificed, tactical reaction time is still of consequence and speeds must be as high as possible consistent with the successful completion of the mission. Some time will be spent in the hover, whilst the ground ahead is scrutinized, and there will also be occasions when a maximum performance dash is required to reduce exposure time across an open area. To take maximum advantage of cover, flight will be at very low level, often only a few feet above the surface, and close to obstacles. The need to maintain a lookout for air and ground activity, to keep contact with other helicopters on the mission, to maintain safe rotor clearance, to navigate accurately and maintain

communications places the maximum strain on the crew.

Although hover-taxy speeds have almost become traditional for NOE flight, a number of operational benefits accrue from the use of higher speeds around minimum power speed; power management becomes easier, fuel consumption is reduced, agility is increased through greater kinetic energy and power/thrust margins, and concealment may be easier through reduced downwash disturbance of dust etc. Handling qualities which would permit the confident use of these higher speeds would enable these benefits to be exploited.

Low 'g' and negative 'g' flight may be encountered during the NOE phase as a result of cyclic push overs and rapid lowering of the collective lever. However, the duration of these manoeuvres should be shorter than during low level and contour flight because of the lower speeds normally being used.

If the maximum advantage is to be taken of full aircraft performance, NOE flight will be characterized by frequent accelerations and decelerations throughout the speed range of hover to maximum dash. In addition to large power and thrust margins the conventional single main and tail rotor configuration requires rapid rotor tilting with associated large pitch attitude changes in order to allow the rapid application of rotor thrust without height gain. Unfortunately, these attitude changes can bring forward view restrictions, increase aircraft exposure, and may undesirably reduce rotor tip clearance. Thrust compounding with a variable pitch propeller may offer benefits, permitting a more rapid acceleration for a given pitch attitude.

Deceleration capability is probably more significant as it represents important avoidance manoeuvres - the traditional "quick-stop", and decelerating turn. The need for rapid deceleration could of course be minimized if NOE speeds were kept very low, but the disadvantages of such a course outweigh the benefits. The operational requirement is for the helicopter to be stopped in the minimum possible distance without loss of concealment. The single main and tail rotor configuration again requires large pitch attitude changes to effect a rearward tilt of the thrust vector, bringing with it tail clearance problems, potentially increased exposure, and possible restrictions to forward view, this time in addition to the rotor speed control problem discussed earlier. Thrust compounding presents a possible solution, using a variable pitch propeller in a braking state; this would assist with the dissipation of main rotor energy, resulting in a better control of rotor speed, and permit higher rates of deceleration at reduced tail down attitudes.

The need to follow the ground track which provides optimum concealment implies changes of direction, as well as height and speed. The demands on aircraft performance and pilot workload in turning flight can be reduced by flying at lower speeds, but this incurs a number of operational disadvantages, already discussed. RAE, UK, piloted simulation studies(1) found that all axes of response were important in turning manoeuvres during NOE flight; pitch and roll for primary manoeuvrability and yaw to maintain balanced turns and counter reaction from torque fluctuations. At 100 kts, roll rates of up to 100°/s were found to be likely whilst at 60 kts on a small triple bend task, very high roll rates up to 150°/s were called for; maximum pitch rates demanded were typically 20-30°/s. The maximum bank angle will normally be approximately 70°, but more may be of benefit to terrain following and concealment. Unobstructed fields of view from the cockpit and precise control are required to give the pilot confidence in using full aircraft performance in this demanding flight regime. Rotor speed control and the intrusion of flight envelope limits will have an influence similar to those discussed earlier under low level and contour flight.

6. THE SUSTAINED HOVER

The requirement for concealment results in the combat helicopter tasks of surveillance, target acquisition, designation, and engagement being made from a concealed hover. The need to achieve adequate fields of view and fire, to aid concealment by minimizing the rotor downwash effects on dust, snow, and leaves, and to enable rapid re-masking below the cover in front by descending vertically, all favour the use of an out of ground effect (OGE) hover.

It must be possible to establish and maintain a stationary hover regardless of wind direction and make rapid and precise changes of heading for target acquisition and weapon aiming. The most demanding conditions for directional control may not occur with the maximum wind strengths, and 'holes' in the yaw control envelope must be guarded against.

The pilot will endeavour to expose only the minimum amount of the helicopter above cover and precise height control, to within $\pm$ 1 ft (300 mm) will be required, especially to gain the maximum benefits from mast mounted sights. It must be possible to acquire rapidly and maintain the pre-launch constraints of any required weapon system. To facilitate the stabilization of target acquisition and weapon aiming systems, vibration levels must be minimized. Weapon system developments such as reduced time of flight will emphasis the need for rapid and precise unmasking and re-masking in order to minimize overall exposure time.

Evasive manoeuvring at the hover will require the ability to move the helicopter rapidly sideways, as well as vertically. A 5% thrust margin is usually deemed

sufficient to hold height and position, but a further margin will be required for agility together with control and response margins.

The demands placed on the pilot in the sustained hover are high, and an automated hover control system should bring a significant reduction in workload, and improvement in accuracy, as well as benefits to survivability.

7. THE ARMED HELICOPTER IN THE AIR TO AIR ROLE

Attack by high performance fighter type aircraft has long been considered and as the primary countermeasure evasive tactics developed and practised. However, the appearance of the Warsaw Pact Hind helicopter with an anti-helicopter capability has brought a new dimension to the battlefield; there will be successors to the Hind, perhaps further specialized for the anti-helicopter mission.

The security conferred by the use of a concealed hover at maximum stand-off is obviously reduced by the ability of the armed Hind to close the gap quickly. It is not all one-sided however, as in closing, the Hind will increase its exposure to our air defence weapons. Because of this, it is likely that it will operate at low level with a consequent restriction in its visual horizon and in its ability to detect its opponent in the concealed hover. The Hind appears to enjoy a speed advantage over current NATO helicopters which it seems likely to retain unless a new generation is produced capable of significantly higher speeds than at present. We are thus currently unable to outrun, or pursue, and in the absence of an effective anti-helicopter weapon, our defence must lie in remaining undetected whilst observing the attacker, ideally continuing with our mission.

However, if discovered and attacked by armed helicopters, an unarmed helicopter must change position to avoid being a 'sitting target', and continue to avoid presenting a valid target as it moves and hides again. Such a defence requires maximum agility in the hover, at low speed, and in a maximum performance dash. The emphasis is once again on remaining down amongst the obstacles which the unarmed helicopter would hope to interpose between itself and the threat, with the requirement for wide unobstructed fields of view from the cockpit. But, given the capability of countering armed helicopter attack with an effective weapon, there is a requirement for target acquisition and weapon aiming implying precise control of attitude and heading. When the need for ultra low flight is disregarded in favour of unrestricted manoeuvring during helicopter versus helicopter engagements, the ability to use both high positive, and negative load factors would be of benefit to both evasion and tracking.

8. NIGHT AND ADVERSE WEATHER

The reduction in external visual cues during flight at night and in adverse weather places added demands on the combat helicopter crew who become more reliant on aircraft systems for safe and successful operation. The need to remain at low level below the radar horizon requires sufficient external reference to fly safely clear of obstacles and pilot night vision aids will be needed. The relatively narrow instantaneous field of view of both passive night goggles (PNG) and turreted Forward Looking Infra Red (FLIR) systems requires continuous head motion by the pilot. Safe PNG flight without stabilization equipment has been demonstrated in a number of aircraft, but comparative experience with the Lynx, fitted with an automatic flight control system, has shown worthwhile improvements in pilot confidence, rate of learning, and control accuracy. The influence of cockpit fields of view on the confidence and ability of the pilot to exploit performance and handling qualities has been emphasised throughout the paper, and it is unlikely that the combat helicopter pilot will ever be able to exploit the full daytime potential of his aircraft whilst flying tactically at night; the combat helicopter with improved handling qualities appropriate to enhanced daytime operation is likely to be capable of more than matching the pilots night flying demands. The need to maintain a sustained precise hover close to obstacles at night is a different matter, and the provision of an effective automatic hover control is likely to be essential.

9. THE TRAINING REQUIREMENT

Defence budgets are likely to remain under pressure, and the costs of operation, as well as the purchase, of combat helicopters are of considerable importance. Safe and easy handling characteristics should lead to shorter training courses, and then enable a suitable level of skills to be maintained with reduced continuation training, as well as keeping down the rate of training accidents. The need to conserve airframe fatigue life may result in flying training being conducted within a reduced flight envelope, with full performance being kept for operations. The pilot would need to be able to transition swiftly to the full capabilities of his helicopter without an increase in the risk of accidents or damage, and good handling qualities extending to the limits of the flight envelope would make a significant contribution.

10. CONCLUDING REMARKS

Concealed low level flight is likely to remain the basis for combat helicopter tactics.

Current combat helicopters are not necessarily optimized for low level, NOE, and hovering flight on the European battlefield.

Improved handling qualities can make a significant contribution to overall combat effectiveness.

"Carefree" handling qualities are required to permit the safe and confident exploitation of full aircraft performance close to the ground and obstacles, including flight at high positive, and negative load factors. Positive cues which draw the pilot's attention to the approaching encroachment of a flight limitation are required to reduce the extra workload and stress induced by the requirement to monitor aircraft instruments during precise and agile flying tasks eg rotor speed transients during maximum manoeuvres.

Handling qualities in need of improvement include flight path control at low workload, particularly with respect to height, during turning manoeuvres at medium to high speed, control of attitude during rapid deceleration, and those relating to directional control at low speed and in the hover. Automated hover control is likely to be essential.

The advent of the armed helicopter in the air to air role emphasises the need for increased agility for evasive manoeuvring while retaining the need for good target tracking capabilities at all speeds.

Safe and easy handling characteristics which are implicit in the reduced workload and enhanced flight envelope should permit a reduction in training costs.

References:

(1) B N Tomlinson, GD Padfield, Piloted Simulation Studies of Helicopter Agility, Vertica VOL 4 No's 2-4, 1980.

FLIGHT EXPERIMENTS WITH INTEGRATED ISOMETRIC SIDE-ARM CONTROLLERS IN A VARIABLE STABILITY HELICOPTER

M. Sinclair
Flight Research Laboratory
National Aeronautical Establishment
National Research Council Canada
Ottawa, Ontario K1A 0R6

SUMMARY

Several flight experiments have been conducted using the NAE Airborne Flight Simulator to investigate the suitability of integrated, multi-axis, isometric controllers for use in helicopters. In these experiments, 3-axis and 4-axis isometric side-arm control configurations were flown successfully through a wide variety of demanding visual flight tasks and a brief instrument flight precision approach evaluation. The experimental tasks, the evaluated controller arrangements and the developed control laws are described, and the results of comparative assessments between isometric side-arm control and conventional control arrangements are presented.

INTRODUCTION

It is inevitable that the micro-electronics revolution should radically alter the interface between the pilot and his aircraft. The invasion of the cockpit is, of course, well underway and the designer is constantly facing a decision — whether to welcome the new technologies as liberating forces or to make a stand in defence of the proven ways. The glass cockpit is gaining wide acceptance; the digital processor is supplanting analog computing methods in many areas of system design and is being applied in new ways to relieve the pilot of tedious tasks; and the glass-fibre optical transmission line is making its bid to replace both electrical and mechanical signalling media.

Along with these changes come a host of "modern" systems which are viable only within the context of a digital, fly-by-wire or fly-by-optics aircraft; it is one of these systems — the integrated, multi-axis side-arm controller — which is the subject of the research programs described in this paper.

In the Fall of 1979 the Flight Research Laboratory undertook a series of experimental evaluations of multi-axis, isometric side-arm controllers in the Laboratory's Airborne Flight Simulator. These early evaluations were partially funded by the Sikorsky Helicopter division of the United Technologies Corporation and grew out of Sikorsky's research studies in preparation for the US Army ADVANCED DIGITAL OPTICAL CONTROL SYSTEM program. Although it was the objective of the initial study to assess these radically new controller arrangements in conjunction with the dynamic and control characteristics of the Black Hawk (UH-60) helicopter, an early decision was made to use the Bell 205A as the basis for the evaluations. This choice had the advantage of eliminating many potentially artificial, simulation-related aspects of the experiments since the Airborne Simulator *is* a variable stability and control Bell 205. The "Huey-like" characteristics of the Simulator provided a relatively simple and well-known baseline against which the unconventional systems could be judged.

These first assessments were simple proof-of-concept experiments designed around pilot evaluations of loosely defined flight-test and operational task sequences. In the intervening months the Flight Research Laboratory has continued to study the influence of multi-axis side-arm control on operational effectiveness and handling qualities, performing direct comparisons between conventional and multi-axis systems in controlled, high-workload flight task sequences. The sample of pilots who have flown the experimental control systems has grown and a relatively extensive base of flight experience has been established with 3-axis and 4-axis integrated, isometric control systems.

From the designer's point of view the appeal of isometric or pressure controllers is evident: they are mechanically simple, light-weight, rugged and compact systems which can readily be incorporated in a side-arm fly-by-wire design. On the other hand, the effects of eliminating the direct control position feedback information — which the pilot would normally obtain from conventional deflection controllers — are potentially far reaching and fundamental. The conventional helicopter control configuration is an integral part of the pattern of control learned by every helicopter pilot and consequently has the status of an international standard. Any benefits gained in a substantial deviation from this arrangement must be weighed against the costs of retraining the pilot's spontaneous control command patterns, particularly in high workload and emergency situations. These trade-offs were at the heart of the flight experiments.

In the sections which follow the controllers and control systems are described and the results of two structured experimental sessions are presented and discussed.

THE EVALUATED SYSTEMS

Hardware

The test aircraft is the NAE Airborne Flight Simulator (Fig. 1 and Ref. 1), an extensively modified Bell Model 205A-1 teetering-rotor helicopter with a full authority fly-by-wire control system at the evaluation pilot's station. The simulator has been equipped with a wide range of motion sensing systems which provide high-quality feedback signals suitable for stability and control augmentation and autostabilization. An on-board hybrid computing system includes a multi-processor, high-speed digital computer which performs signal conditioning, control system implementation and simulation "modelling" — the essential tasks of simulation.

Two isometric controllers and supporting side-arms have been fitted to the structure of a standard Bell 205 crew seat. During the first of the two experiments described in this paper, the seat was configured as shown in Figure 2 using off-the-shelf, commercial controller units and hand grips. For later experiments, and specifically for the second experiment described below, an adjustable arm structure was developed and new grips which more nearly conform to the shape of the cupped left or right hand were cast (Figs. 3 and 4). The force sensing and transducing functions of the handles, in both of these installed configurations, were performed by four piezo-resistive strain gauge units which produce electrical outputs for four independent commands — fore and aft, left and right, up and down forces, and torque about the vertical axis (Fig. 5). The nominal value of command to electrical output sensitivity for each of these channels is shown below.

Command	Sensitivity	Maximum Output
Left/Right	0.5 volt/lb	10 volts (linear)
Fore/Aft	0.5 volt/lb	10 volts (linear)
Up/Down	0.25 volt/lb	10 volts (linear)
Clockwise/Counter-clockwise	0.167 volt/in.-lb	10 volts (linear)

The handles are elastically very stiff but not rigid and their supporting structures increase the compliance of the complete system to forces and moments applied at the hand grip; nevertheless the overall deflections are small and do not provide *position* feedback to the pilot in the form of discernable force-deflection relationships. To compensate for this reduction in system feedback information a control actuator position indicator was added to the experimental system late in the evaluation phase of the first experiment. The indicator shown in Figure 6 mounted above the instrument panel in the evaluation pilot's forward field of view, provided easily interpreted information concerning tip-path-plane orientation and tail rotor collective actuator positions.

Software

The command and logic signals from each handle and a force-proportional signal from the evaluation pilot's pedals were sampled and sent to the Simulator's master digital computer where channel assignments, mode selection, control law and stability augmentation calculations were performed. (A typical control channel is shown schematically in Figure 7.) Channel assignments for the five control configurations tested in the first experiment, including the two "primary configurations" which have been the subject of more recent investigations, are depicted in Figure 8.

It is clearly necessary to *trim* the isometric control systems, just as it is necessary to trim deflection controllers which have force feel gradients. System demands for a steady-state control input must be met by adjustments to the command signal datum — not by a continuous force applied by the pilot. In a sophisticated fly-by-wire flight control system this trimming function may be accomplished through the mechanism of high-gain command control laws which are inherently self-trimming. For example, a form of rate command/attitude hold mode in roll, pitch and yaw with *acceleration command* in the vertical axis would obviate the need for manual trimming. However, the experiments conducted at the Flight Research Laboratory have emphasized simple, low-gain feedback augmentation and control for which trimming *is* necessary and the chosen approach is depicted in Figure 7. An integrating path is added in parallel with the proportional signal from the pilot's input command. When the "integral-trim" switch is closed the integrating path produces a relatively rapid trim follow-up signal in response to any non-zero command. The evaluation of various trimming schemes and the development of this integral trim system are discussed in some detail in Reference 2.

Finally, the control law software incorporated two levels of stability and control augmentation or autostabilization in addition to a "direct-drive" mode. These were:

a. rate command/attitude hold in roll and pitch with augmented yaw rate damping

b. augmented roll, pitch and yaw rate damping

c. direct-drive (the Bell 205A-1 with stabilizer bar removed and horizontal stabilizer fixed)

THE FIRST EXPERIMENT

The preliminary series of flight tests comprised a development test phase during which suitable control signal shaping and integral trimming gains were established, followed by formal evaluations of the two primary control configurations — the 3-axis and 4-axis systems depicted in Figure 8. (Five engineering test pilots participated in these flight evaluations.) The system designated "3-AXIS/TWIST COLLECTIVE" was flown by several pilots and thoroughly evaluated by one, and the other channel assignments shown in Figure 8 were flown only briefly to assess their value as training aids. The evaluation tasks were chosen from a set of loosely defined flight test and operationally oriented manoeuvres including take-off and landing, hover manoeuvring, circuits, nap-of-the-earth flight, precision instrument approaches and slope operations.

A brief summary of the significant observations and conclusions from these first flights of a helicopter using integrated, multi-axis side-arm control is given here to provide a context for presentation of the more recent experiments. Reference 2 provides more detailed results.

In general it was found that pilots adapted with surprising ease and speed to the use of multi-axis isometric control — each *of the evaluators flew the helicopter confidently, using the 3-axis or 4-axis system, after a very brief familiarization flight.* (These comments apply for the augmented and autostabilized helicopter control modes. Although the pilots were able to perform all of the manoeuvres and task sequences satisfactorily in the "direct-drive" mode, using either of the primary control configurations, the workload was undesirably high in the more demanding tasks.)

Multi-Axis Control

The experiment showed that a helicopter can be flown through a wide range of visual and instrument flight tasks using either a 3-axis or 4-axis integrated isometric side-arm controller — within the bounds of normal helicopter workload demands.

Control Position Feedback

An adequate level of control position feed back may be provided by a well-designed control position indicator (CPI). Using the rudimentary CPI shown in Figure 6 the pilots were able, for example, to perform slope landings and take-offs — manoeuvres which require knowledge of tip path plane orientation. For visual flight manoeuvring tasks where control authority remaining or absolute control position is not a primary concern, the force feedback alone was sufficient.

Left or Right Hand Control

Although left-hand operation was not emphasized in the test planning, several of the pilots developed the skill to the point where they could perform all of the experimental tasks adequately using the left-hand controller.

An "ambidextrous", multi-axis control system could have a fundamental influence on cockpit layout since it would allow the pilot to perform auxiliary manual tasks with either hand — interacting with systems on either side of his seat position.

The Secondary Configurations

The "3-AXIS/TWIST COLLECTIVE" configuration could be mastered and was flown successfully, but it was prone to applications of collective control in the wrong sense and was considered markedly inferior to the primary 3-axis version which translated vertical forces as collective commands.

The two-handed configurations, both of which controlled roll and pitch with the right-hand, heave with the left-hand and yaw with pedals, were not systematically evaluated in this program. Each could be flown with ease.

THE SECOND EXPERIMENT

The qualitative nature of the first series of tests left some important questions unanswered: whether, for example, pilots can learn to perform very demanding manoeuvring and control tasks using the multi-axis isometric controllers — with precision and ease comparable to their performance using conventional controls; and, whether the patterns of control — the characteristics of the command signals which the pilot uses to meet specific manoeuvring or stabilizing control requirements — differ fundamentally for the two types of control.

To gain some insight into these questions, a second flight test session was designed around a structured series of low-altitude manoeuvring and precision-control tasks. The test course is depicted in Figure 9 which shows the nature of the flight segments and the layout of ground markers used to guide the pilot from task to task. For example, on the accelerate-stop leg the pilot attempted to pass through the "gate" at a ground speed of 40 knots and to stop over the designated point at the end of the run — maintaining a uniform height above ground; the landing was to include a smooth approach followed by a touchdown on a marker and on a preassigned heading; the lateral translation was to follow the ground track, maintaining constant heading, height and lateral velocity.

The tasks made combinations of demands which required co-ordinated cyclic, main rotor collective and tail rotor collective commands — in some instances to decouple the helicopter motions and in others to induce the co-ordinated, multi-axis manoeuvres.

Each subject pilot was trained to a high level of proficiency in executing the course with conventional controls and with one of the primary isometric controller configurations. He then flew the course eight times in a single flight, alternating pairs of runs first with conventional controls, then with the multi-axis configuration. On a second and similar evaluation flight, eight more runs were completed, this time beginning with a pair of runs using the hand controller. Following a short "deprogramming" period and a period of retraining with the other of the two primary controller configurations, the procedure was repeated so that both the 3-axis and 4-axis isometric configurations were compared directly with conventional stick, collective and pedal control.

Four pilots completed the full test sequence, two engineering test pilots who had participated in the earlier multi-axis controller study and two pilots who had only very brief exposure to isometric control prior to this comparative study.

All of these test flights were conducted using the rate-augmented control mode, the basic simulator with augmented rate damping in roll, pitch and yaw. The 3-axis and 4-axis isometric configurations employed continuous integral trimming in all control axes with control sensitivities and integral trim rates individually adjusted but generally close to the values established in the earlier tests. For the conventional control system the sensitivities were set equal to the Bell 205 values and control forces simulated a boosted control system without artificial gradients.

RESULTS OF THE SECOND EXPERIMENT

Four distinct types of data and information were collected during the test flights and following the completion of the experiment. Time histories of control forces, control displacements where appropriate, and of aircraft state variables were recorded on the simulator's data acquisition system. Touchdown accuracy and lateral tracking accuracy were measured using sighting transits. Each segment of each run was described qualitatively in brief notes compiled by a ground observer, and finally the pilots were asked to complete a brief questionnaire which addressed the relative difficulty experienced and precision attained with the systems being compared.

The time histories are being analyzed to determine similarities or differences in patterns of control and precision of control between runs using integrated isometric systems and those using conventional displacement control. For example, the heave and yaw command signals consistently exhibit significant content at higher frequencies when these channels are controlled with the force handle. On the other hand the ground speed, pitch attitude and longitudinal cyclic control* time histories for the accelerate/stop segments show no apparent distinguishing features which separate multi-axis isometric from conventional control. Although the pilot's actions with the two types of controllers are fundamentally different, the demands which he makes on the cyclic actuators and the resulting manoeuvring performance are similar.

---

* A valid control demand comparison can only be made at the point where signals are being sent to the helicopter control actuators. The originating "commands" are fundamentally different, being forces in the one case, displacements in the other.

Landing precision is illustrated in Figure 10 in the form of touchdown dispersion from the target point. It can be seen that performance with the conventional controls was superior to that with the multi-axis systems and that the dispersion patterns for the two cases were distinctly different. This result should be anticipated simply on the basis of the higher level of experience of the evaluation pilots with deflection control systems. The difficulty *does* appear to be peculiar to control in the last few inches of the landing approach however, since steady station-keeping over a ground target, at skid heights of a few feet, could be performed very precisely with the integrated isometric controllers. (Opinions differ concerning the reason for the forward-starboard drift from the target during the side-arm controlled touchdowns. It should be recalled that the evaluation pilot occupies the right-hand seat of the simulator and, as a consequence, the forward-starboard field of view is essentially unobstructed.)

Although the objective data are in themselves of interest, the primary reason for specifying the tasks precisely, and for measuring performance, was to constrain the pilots to a consistent test procedure for all of the evaluated systems. Subjective, comparative assessments were considered to be the most important output of the experiment. This subjective information, which was supplied by the evaluators in response to a brief questionnaire, provides some insight into the process of adaptation to the unconventional, integrated control systems. For each segment of the course the pilots were asked to characterize the relative difficulty which they experienced in performing the task and the relative precision with which they considered they accomplished the task, comparing the two control systems used on the preceding flight. A consolidated picture of these opinions is presented in Figure 11 where all of the responses relating to all of the course segments have been collected. There is no rigorous analytical basis for this *linear addition* of the individual responses — the dividing line between "more difficult" and "much more difficult" was subjectively and individually defined — nevertheless the composite picture has qualitative significance. Each assessment necessitated, first and foremost, a simple decision concerning which of two recently performed tasks was easier to perform and which was performed with greater precision. These figures underline the message which pervades the general comments of the subject pilots: for very demanding manoeuvring control tasks, the more familiar conventional control system is marginally superior — in both precision and ease of operation — to the integrated isometric controller arrangements. As the level of experience with force control increases however, the differences in workload and performance between conventional and isometric control diminishes. In these direct comparisons with conventional controls it was also evident that the 3-axis isometric controller mode was considered slightly superior to the fully integrated 4-axis mode, although the difference was judged to be small and perhaps of no operational significance.

CONCLUDING REMARKS

Force control inputs have been widely used in airplanes both as quickening devices for deflection control systems and more recently as primary control signals in advanced digital flight control systems. The experiments described herein are concerned not just with the transfer of *these* control methods to the helicopter environment but with the added dimension of integrated, multi-axis control — an addition which renders the systems different in kind as well as degree. The helicopter flight experiments have shown that multi-axis isometric side-arm control systems can be used successfully to perform a wide variety of demanding flight control tasks.

In recent weeks a new multi-axis *force* controller, similar to the 4-axis system described above, but with much greater compliance, has been installed in the side-arm seat and flown in the Airborne Flight Simulator. The deflections of this handle in response to applied forces and moments are still relatively small — only one half inch of displacement at the mid-hand position for a 20 lb pitch or roll command, for example — but the resulting force/deflection characteristics are discernible and appear to provide useful feedback information to the pilot. This system will now be evaluated systematically and compared with the isometric controller and conventional deflection control systems.

Finally, it should be emphasized that these evaluations have focused upon relatively simple control laws with low-gain feedback stability augmentation. If integrated, multi-axis side-arm controllers are adopted for an operational helicopter, they will undoubtedly be part of a very *sophisticated* control system incorporating electric or optical signal transmission and digital control-law computation. In such a control environment, task and mission optimization of command control laws is not only possible but practical, and therefore a full assessment of the controllers should include thorough investigations of these advanced control laws and concepts. Future phases of the Flight Research Laboratory's program will address control law optimization and the implications of these extraordinary control systems for the helicopter certification process.

REFERENCES

1. Sinclair, S.R.M.; Roderick, W.E.B.; Lum, K. — *The NAE Airborne V/STOL Simulator.* Proceedings of the AGARD Flight Mechanics Panel Symposium on Rotorcraft Design. AGARD CP 223, 1975.

2. Sinclair, M.; Morgan, M. — *An Investigation of Multi-Axis Isometric Side-Arm Controllers in a Variable Stability Helicopter.* NRC, National Aeronautical Establishment Report LR-606, National Research Council Canada, Ottawa, August 1981.

FIG. 1: THE NAE AIRBORNE FLIGHT SIMULATOR

FIG. 2: SIDE-ARMS AND CONTROLLERS INSTALLED (FIRST EXPERIMENT)

FIG. 3: SIDE-ARM SEAT AND CONTROLLERS ARRANGEMENT FOR SECOND EXPERIMENT

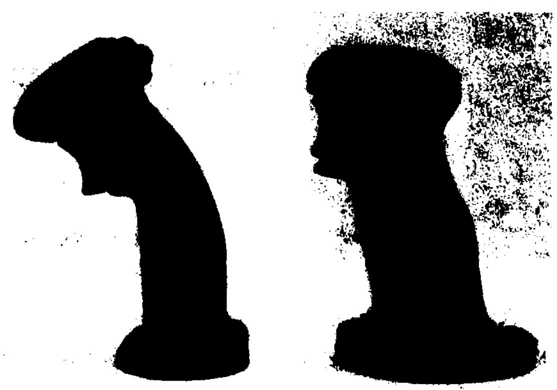


FIG. 4: OLD AND NEW HAND GRIPS

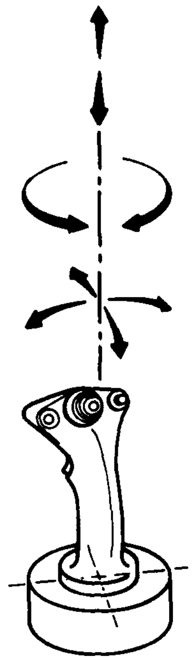


FIG. 5: FORCE AND MOMENT SENSING AXES

FIG. 6: CONTROL ACTUATOR POSITION INDICATOR

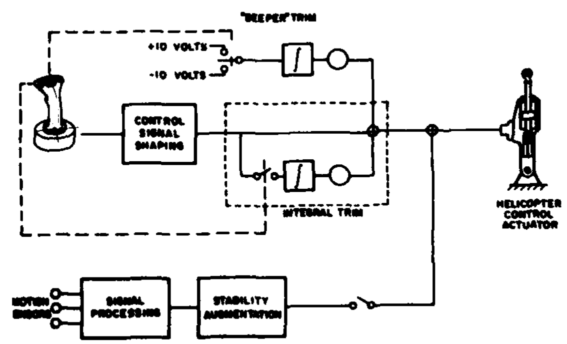


FIG. 7: CONTROL CHANNEL SCHEMATIC

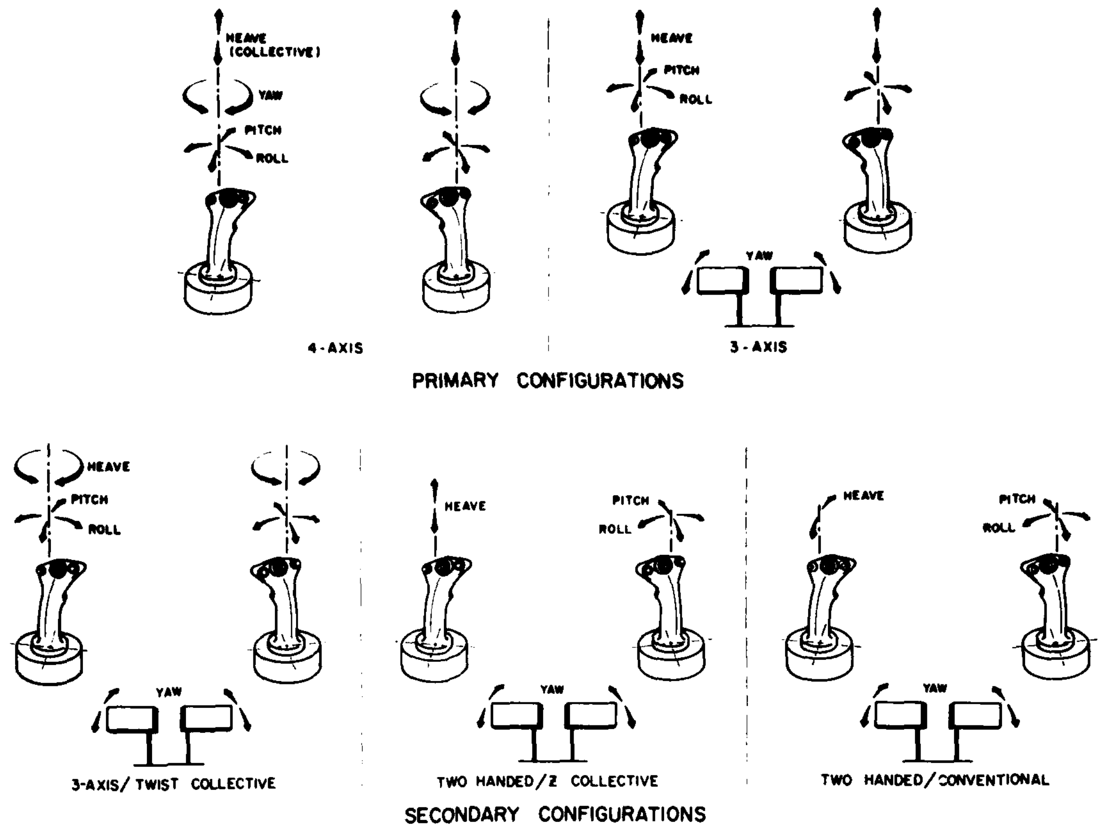


FIG. 8: CONTROL CHANNEL ASSIGNMENTS FOR EVALUATED MODES

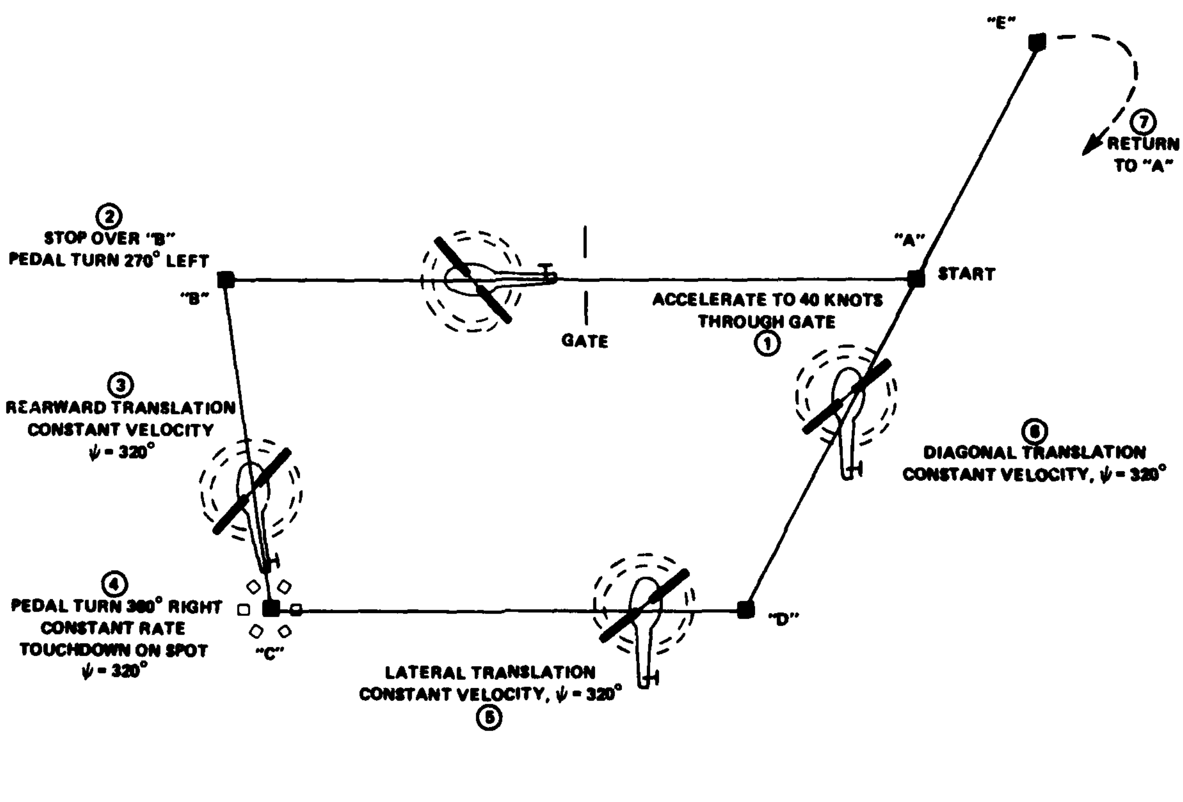


FIG. 9: MANOEUVRING COURSE

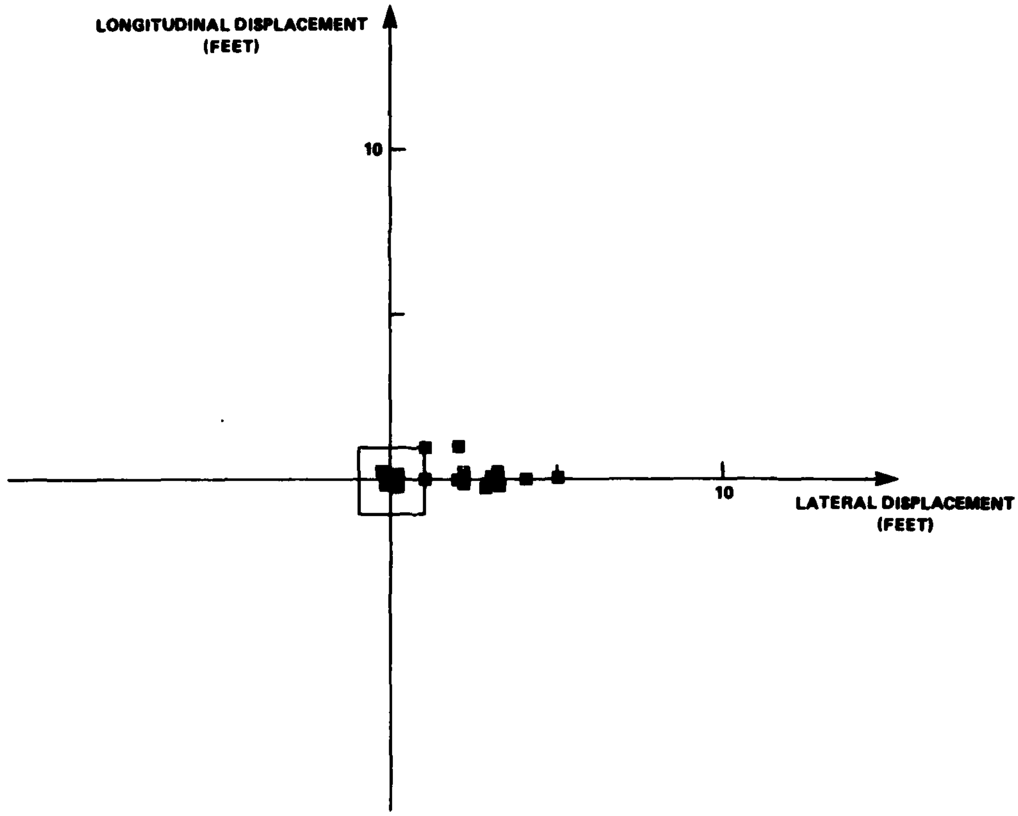


FIG. 10(a): TOUCHDOWN PRECISION – CONVENTIONAL CONTROLS

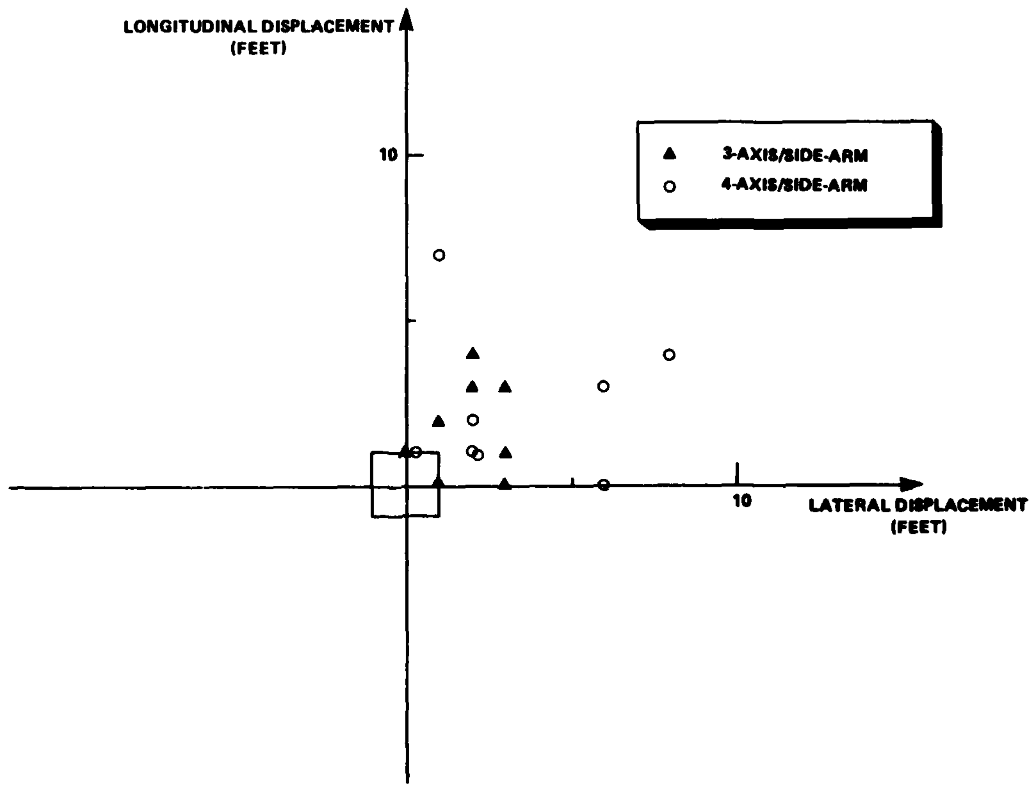


FIG. 10(b): TOUCHDOWN PRECISION – SIDE-ARM CONTROLLERS

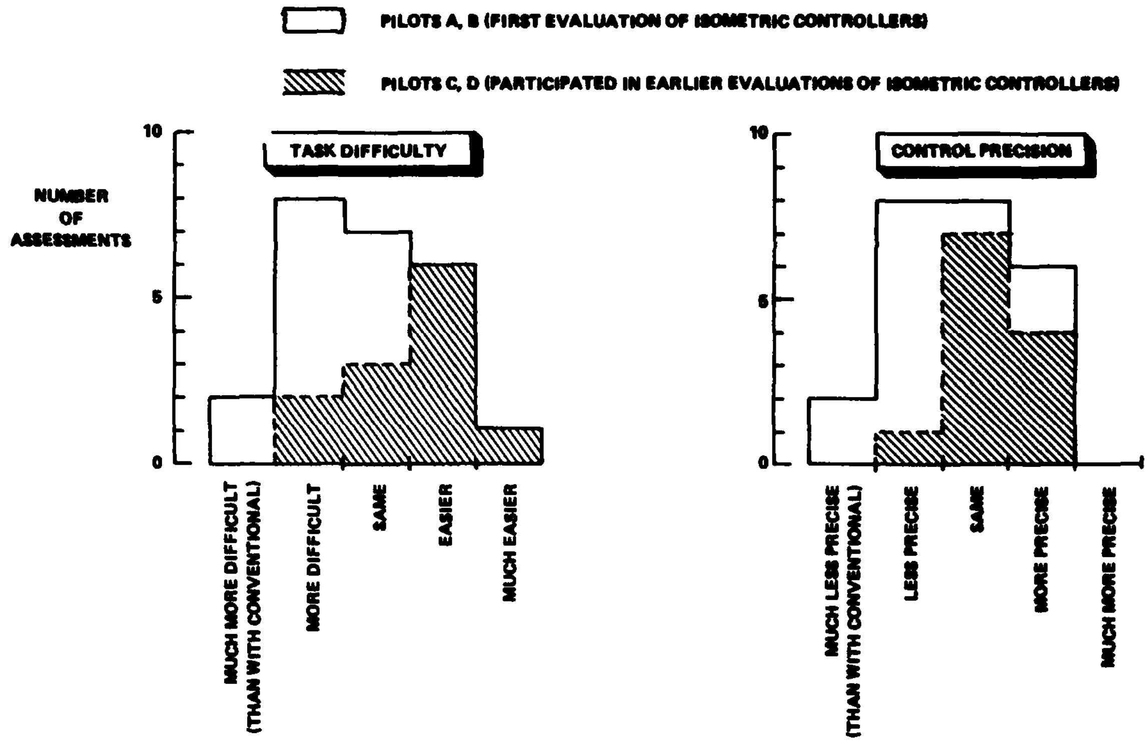


FIG. 11(a): SUBJECTIVE COMPARISONS OF 3-AXIS ISOMETRIC CONTROLLER WITH CONVENTIONAL CONTROLS

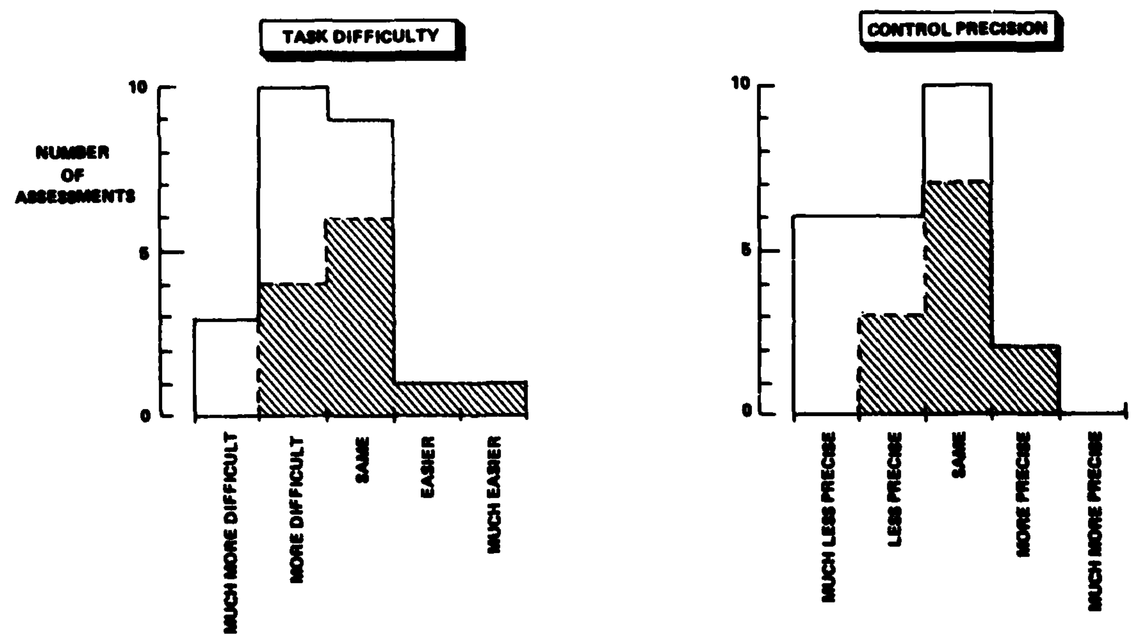


FIG. 11(b): SUBJECTIVE COMPARISONS OF 4-AXIS ISOMETRIC CONTROLLER WITH CONVENTIONAL CONTROLS

STABILITY AND CONTROL FOR HIGH ANGLE OF ATTACK MANEUVERING.

by
W. Kraus, H. Przibilla, U. Haux
MBB Flugzeuge, Postfach 801160,
8000 München 80, West Germany

SUMMARY

On a delta canard configuration an optimum division of control devices for maximum control power at high angle of attack (A.O.A.) is shown and a special-trim schedule gives best directional and lateral stability in this flight regime.

This aircraft configuration was used for an air to air combat simulation. The control system lay-out at high incidence included thrust vectoring in pitch and yaw to support the aerodynamic control surfaces. Simulation results in terms of rates and accelerations in pitch, roll and yaw axis for a set of different poststall (PST) maneuvers are shown to prove that the aircraft is controlable and that active tactical maneuvers can be flown in this flight region.

1. INTRODUCTION

This paper is concerned with the possibilities of maneuvering a fighter aircraft at and beyond maximum lift. The tactical value of such capability in Short Range (SR) Air Combat has been the subject of various analytical studies, trajectory optimizations, computerized and manned air combat simulations, see references [1, 2]. It was found, that air combat capability can be improved by means of suitable high A.O.A. maneuvers to a degree which is unachieveable by α limited maneuvers at any possible level of energy performance beyond that of contemporary fighter aircraft.

There is a significant change of air combat characteristics to be expected from the use of all aspect SR weapons (see reference [3]) in terms of a greater importance of instanteneous lower speed maneuvers. Frontal engagements will prevail and thus combat success is dependant much more on the ability to achieve an earlier firing solution even at the expence of energy. Energy will often be sacrificed for positional advantage. PST-maneuvers - if properly conducted - offer a turn radius advantage over the conventional opponent, which may lead to that positional advantage, if:

- thrust to weight ratio is higher than 1,
- sufficient control power is available at associated low speeds and A.O.A.s up to 70°.

Suitable conducted PST maneuvers are characterized by

- 3 dim. trajectories
- short duration (lower than 7 seconds)
- high pitch rates and high yaw rates
- marginal angles of sideslip (theoretically none)
- rapid change of speed and flight conditions.

PST maneuvers do not constitute a special mode of flight. They are merely an extension of well known instantaneous low speed maneuvers. They can easily be adopted by the normal pilot, who however, has to get used to the condition of rolling around the velocity vector in order to avoid excessive angles of side slip. Note that in most tactical situations PST-maneuvers are used to achieve a positional advantage and the fire opportunity will occur after the return into the normal A.O.A. regime. Very seldom PST capability is being used for weapon pointing.

The paper investigates the requirements for handling and controlling a particular fighter design in the tactical PST flight regime. Results are based on wind tunnel tests and simulations of typical PST maneuvers. It was the aim to develop the technical means to achieve the level of controlleability which was found to be tactically necessary in the preceeding tactical air combat simulations.

Fig. 1 shows this level. So we had an aiming goal for designing the aircraft and its controls. It was soon very clear, that the desired angular rate onsets in high A.O.A. regime for tactical maneuvers only could be fulfilled if additionally to the aerodynamic controls a speed independant momentum control system, a vectored nozzle for pitch and yaw controls was used.

The aircraft thereby should have in this flight regime the required maneuverability level. To reach these goals, stability and control requirements for tactical high A.O.A. operation have to be established.

Doing the flight system layout, manned simulator studies are very helpful. The handling qualities and maneuverability performance can be adapted to the established stability

and control requirements at maneuver relevant control inputs. Then with a 'pilot in the loop' simulation on a fixed base cockpit station, the system can be rated and standard PST maneuver types can be used, to qualify the system performance.

2. DESCRIPTION OF CONFIGURATION AND AERODYNAMIC LAYOUT

Next a short description of the configuration and its aerodynamic control devices and their optimum use in high A.O.A. is given. A detailed description of the influence of configuration components and control settings has been given in [4].

The simulated aircraft is a Delta-Canard configuration (see Fig. 2) with twin vertical tails, 4 leading edge (L.E.) and 4 trailing edge (T.E.) flaps. The canard off configuration shows neutral stability in the pitching moment up to A.O.A. ≈ 8°. The fixed canard contributes a destabilization of about 7% in this A.O.A.-regime. Canard on and canard off longitudinal stability are well balanced in order to reduce trim drag by the spanwise downward deflection of the T.E. flaps and on the other hand to achieve a restabilization of the aircraft by releasing the canard in the case of a CCV-system failure. Canard and outer T.E. flaps are used for pitch control and stability augmentation. Roll control is superposed to the trim and pitch control deflection of the outer T.E. flaps. Yaw control is done with the 2 rudders of the twin fin.

As canard and T.E. flaps settings both influence pitching moment, longitudinal and lateral stability and control authority, it is necessary to find a trim schedule dependant on A.O.A. and Mach number which is optimized under the following aspects:

- minimum drag in the conventional A.O.A.-regime
- sufficient control authority at high A.O.A.
  - sufficient pitch down acceleration rate
  - reduction of lateral and directional instability in the critical A.O.A.-regime (30° - 50°)
  - quick response by aileron deflection for roll stabilization
- balanced loads and actuation rates of control devices.

The canard is held in a nearly 'no load' position for medium A.O.A. to keep the strong vortex interaction between canard and wing low because the interaction and their asymmetric break down at sideslip angles is responsible for lateral instabilities.

The L.E. flaps are deflected downwards also to improve lateral stability.

The T.E. flaps are deflected downwards at low A.O.A. because of performance reasons. At medium A.O.A. the trim schedule shows a separation of outer and inner T.E. flap setting. The aileron efficiency sharply drops at A.O.A. of about 30° to about 1/3 of the low A.O.A.-region and the downward deflected aileron only takes a 20% share of this capacity, see Fig. 3.

Due to this effect a better roll response is experienced by a more upward basic setting of the outer T.E. flaps. This schedule allows sufficient roll control beyond aircraft stalling with conventional ailerons.

The trim schedule of Fig. 2 compromises these optimization aspects and has been applied in the high A.O.A. simulation. Lateral stability parameters associated with this trim schedule are shown in Fig. 4. Though there is a region of directional instability, the spin departure parameter $Cn_{\beta dyn}$ is kept positive throughout all A.O.A.

Fig. 5 illustrates once more the spin departure parameter $Cn_{\beta dyn}$ together with four levels of spin pronity dependant on sign and value.

As to see, the basic configuration can show severe and abrupt yaw departure with large sideslip excursions between 15° and 40° A.O.A., if the L.E. flaps are not deflected. At trim conditions, however, as already shown before, $Cn_{\beta dyn}$ remains positive in the critical α-region, becoming even more positive at very high A.O.A. due to the stabilizing dihedral Cl_β.

A more significant criteria for proverse/adverse sideslip build up can be found in the frequency ratio, $\omega\phi^2/\omega\beta^2$ = LCDP $\cdot$ $\cos\alpha/Cn_{\beta dyn}$. Values greater 1 indicate proverse sideslip (p and β having different sign) and values lower 1 inverse sideslip (loss of roll power due to sideslip build up).

For values lower 0, roll reversal may be expected, and the roll acceleration is inversed to the demanded roll stick direction. The frequency ratio is plotted versus A.O.A. for trim conditions and zero trim sideslip. For α > 35°, the aircraft shows a weak tendency toward roll reversal. This tendency becomes more intensive for out of trim conditions.

For roll control a schedular was established, depending on A.O.A. and Mach number. Fig. 6 shows the resulting aileron effectiveness versus A.O.A. for the remaining roll authority relative to the pitch control demands on the T.E. flaps.

Fig. 7 shows the rudder effectiveness. It decreases markedly at A.O.A. greater than 30°. At about 45° A.O.A. the rudders become totally ineffective.

3. CONTROL SYSTEM LAYOUT AND SIMULATION RESULTS

Having analyzed the basic airframe stability behaviour and the available control power in the low speed regime, the next step was to select appropriate control laws and feedback signals and to show in form of a 6 degree of freedom (DOF) simulation study, that the presented configuration can remain under control up to very high angles of attack with the implementation of thrust support by vectoring both engine nozzles with a given control authority of 10°. The main demands regarding stability and control requirements for tactical high incidence maneuvering can be summed up as follows:

- prevention of unrecoverable pitch departure at fast pull-up maneuvers, caused by insufficient nose-down control power at higher angles of attack and low dynamic pressure levels;
- prevention of spin departure due to yaw or roll divergence, caused by negative directional flight path stability or roll reversal;
- sufficient control power in all three axis to maintain the above mentioned stability performance and maneuverability level for tactical relevant PST-operations.

Beside the already mentioned stability problems in the lower to medium A.O.A.-region, kinematic and inertial coupling effects become dominant on the aircraft motion at medium angles of attack. These effects show up in kinematic α-β-exchange during high incidence body axis rolls, as well as in pitch-up and proverse or adverse yawing moments generation due to roll/yaw or pitch/roll coupling.

To determine the necessary control power in all axis, these effects had to be considered for the Command & Stability Augmentation System (CSAS) lay-out.

The next figure (Fig. 8) shows the finally selected control laws and feedback signals for the pitch axis. The main features can be summed up as follows: The stick force per g depends nonlinear on the stick deflection range with the calibrated dynamic pressure as a parameter. Having reached the maximum g-load, a break in the stick force gradient indicates to the pilot, that he will enter the PST-region by pulling beyond $\alpha C_{L_{max}}$, now controlling angle of attack. The command signal is fed to the canard and to the T.E. flaps with different authorities, dependant on angle of attack and dynamic pressure. Added downstream of the feedback signals α and $\dot{\alpha}$, are the trim values for the canard and the flaps with schedules described before. In the canard loop, an α-dependant schedule limits the nose-up canard authority for pitch departure prevention at fast pull-up maneuvers.

Demanding for a fast and precise controlable aircraft at high incidences, the stability loops were designed for a constant natural frequency of 3 rad/sec and for a damping ratio of 0.7. These short period response requirements correspond with the MIL 8785B specification of medium nz_α-levels. Without Vector Nozzle Control (VNC) in the pitch axis, the mentioned ω_n, ζ-values could not be reached. Without thrust support, heavy α-overshoots and even pitch departures resulted at lower dynamic pressure levels, when pulling α-rates greater than about 25°/sec.

Thereby the canard saturated for angles of attack greater than about 40°. Therefore VNC was implemented in an early stage of the simulation study. The VNC-loop was automatically phased in as a function of angle of attack and calibrated dynamic pressure. The switch lets through the strongest signal of both. The phasing in values were varied in the simulation session and finally fixed for this configuration at $\alpha^* = 20°$ and $\bar{q}^* = 3\ KN/m^2$.

The Fig. 9 shows a simulation result with VNC and "Max. Dry" thrust setting. Here we can see the resulting maximum attainable α-rates and the needed control deflections in relation to the trim settings and authority limits, for fast pull-up maneuvers up to 70° angle of attack. Also shown in this figure are the maximum deflection rates of all three pitch moment generators. The next figure (Fig. 10) shows a time history for rapid pull-up maneuvers at a thrust setting of "Max. Dry". The maneuver was started at M = 0.6 for an altitude of 20 KFT. The aircraft was rolled to an initial bank of about 40° entering a turn. Thereby the aircraft was pointed away from the flight path by fast pitch stick commands of different authority, reaching maximum angle of attack values up to 70°, at a dynamic pressure level of 2 KN/m². The maximum attainable pitch rate was about 35°/sec for this thrust setting, with an angular rate onset of 50°/sec². For angles of attack greater than 45 to 50°, the canard saturated. But the additional pitch control power of the nozzle prevented the aircraft from pitch departure, allowing enough nose-down moment generation, to recover from these deep stall conditions. The maximum possible deflection rate of 70°/sec was not reached.

The next Fig. 11 shows the selected control laws and feedback signals for the lateral/directional axis. The layout of these loops was dominated by the demand of absolute spin departure resistance for maneuvering conditions. As already said, both spin departure parameters are strongly negative for out-of-trim conditions due to detrimental interference effects of the dynamically deflected canard and T.E. flaps on both the stability derivatives Cl_β and Cn_β.

To minimize kinematic sideslip build up at roll control inputs for higher angles of attack, it showed to be mandatory to roll around the stability axis instead of the body axis, thus preventing α-β-exchange. As shown in the figure, the lateral stick inputs, scaled for stability axis roll rate demand, produce primarily body axis roll rates at lower angles of attack and mainly body axis yaw rate at higher angles of attack. Thus implementing a roll stick-to-rudder-interconnect with the control law given by $r_{Demand} = p_{Demand} \cdot \tan\alpha$ means, that the aircraft is forced to roll around the velocity vector or around the flight path. It is most obvious, that the resulting conical motion of the aircraft around the flight path, eliminates the kinematic α-β-coupling. At very high A.O.A., the aircraft can not be rolled anymore around its body axis, but can be sliced around the yaw axis.

To guarantee a departure free aircraft, a high augmentation β-feedback was provided beside the Roll Stick Rudder Interconnect (LSRI) concept, to stiffen the flight path directional stability $C_{n\beta_{dyn}}$. The β-feedback loop was designed for a natural dutch roll frequency of 3 rad/sec in the whole α-region with a likewise constant damping ratio of 0.7. This β-response characteristic was sufficient to prevent yaw departure under all maneuver conditions and roll reversals, ending in a spin entry condition. The gains in the stability loops were calculated with reference to the $C_{n\beta_{dyn}}$-values for trim conditions, shown before. To sustain the aircraft to roll around the velocity vector at out-of-trim conditions at dynamic sideslip build-up, a stability axis yaw damper was implemented. All gains were α- and $\bar{q}$-compensated.

Simulating the system without thrust support in the yaw axis showed, that the rudders could be saturated already at medium angles of attack for dynamic pressure values less than about 5 KN/m^2. For angles of attack greater about 35°, no relevant yaw rates could be maintained. Therefore, as in the pitch axis, the rudders were supported or substituted by the vectoring engine nozzles, generating body fixed yaw acceleration, dependant on thrust setting, Mach number and altitude.

As in the Aerodynamic Surface Control (ASC) loop, for the VNC a similar rollstick-to-yaw nozzle-interconnect was designed, and the same feedback signals were used. Thereby the pitch and yaw axis have the same authority status with an overall authority of 10°.

The next figure (Fig. 12) shows the resulting roll/yaw rate performance for three thrust settings: "idle", "Max. Dry" and "Max. Reheat", as well as "without VNC". The values resulted at full roll stick inputs for the different α_{trim}-settings. Without VNC, no relevant yaw rate can be attained for $\alpha > 35°$. Thereby β-excursions > 10° could be observed at fast roll stick reversals, with the rudders in a limit cycle. With VNC and a minimum thrust setting of "Max. Dry", the sideslip excursions at fast roll stick reversals and at cross-control inputs could be held less than 1°. The next figure (Fig. 13) shows a roll stick reversal maneuver at an angle of attack of about 65°. Here we can see the fast yaw rate build up with about 30°/sec^2 yaw onset in a full stick reversal.

Having demonstrated in a first simulation session, that the present delta-canard configuration remains controllable with the selected control laws and the implementation of Vector Nozzle Control in the pitch and yaw axis an operational pilot was asked to rate the whole system, flying different maneuver types with the standard task of a 180° heading change without pedal inputs, thereby involving high angles of attack up to 70°. The shown maneuvers were flown without target display. Therefore the pilot could only orientate on the following instruments:

- ADI (attitude director indicator) - altimeter
- HSI (horizontal situation indicator) - speedmeter
- g-meter
- α & β indicator - vertical velocity indicator.

All here shown maneuvers were flown with phased in Vector Nozzle Control at constant thrust settings for the initial conditions of M = 0.6, 15 kft altitude and 1 g level flight.

The next figure (Fig. 14) shows the 180° heading change performance in a horizontal turn with an initial bank of 90° and "Max. Dry" thrust setting. Having pulled to 8.5 g, reached at about $\alpha_{C_{Lmax}}$ the pilot did not hold the maximum turn rate, but pulled into the PST-regime, reaching for a short elapsed maneuver time a maximum angle of attack of about 65°, whereby the speed dropped to 160 knts. The 180° heading change was reached at α_{max} after 8 seconds, thus having an average turn rate of 23°/sec. At the lowest speed level, the flight path was vectored to about -50°. During PST-entry the maximum pitch rate was about 35°/sec. The maneuver was ended by a fast α-recovery to about zero degree and a simultaneous roll to level flight.

The next figure (Fig. 15) shows the 180° heading change, flown in the vertical with zero degree initial bank. Again the max. attainable normal load factor was reached after 2.5 sec. and pulling over the top with "Max. Reheat" throttle setting the inverted attitude was reached within about 10 seconds. The minimum speed in this maneuver was about 80 knts at a maximum A.O.A. of 55°.

The next figure (Fig. 16) shows a more PST typical flight path change. The so-called PST-Slice was initiated at a constant angle of attack of about 70°, obtained at full aft stick. Here we see, how the aircraft is forced to roll around the flight path with a settled flight path angle of about -70° in a slightly nose down position during several 180° heading changes with an established yaw rate of 35°/sec, ending the 180° pointing in about 6 seconds. The slice around the near vertical velocity vector, with an all-aspect gun pointing capability, minimizes the aircrafts combat maneuvering range, thereby developing sink rates greater than 6000 ft/min. The slice can be stopped fast and with a good settling time.

The last figure (Fig. 17) shows the angular rate onsets in body axis system, extracted from the pilot simulation with the actual system in the before described maneuvers. The maximum demanded values in pitch, roll and yaw axis are referenced to the ASC only levels and to the additional VNC potential acceleration values. As shown, the nose-up control power of the canard and T.E. flaps was not needed for PST-entry, but was not sufficient enough to pull the aircraft to the maximum useable angle of attack. Using the additional thrust support in the pitch axis, the objectives for air-to-air simulations were reached at higher angles of attacks. For a fast α-recovery and for preventing pitch departure during fast pull-ups, the nose-down ASC-power is not sufficient at lower speeds and has to be sustained by the Vector Nozzle with "Max. Dry" as a minimum thrust setting. The roll control power of the ailerons with α-dependant authority, showed to satisfy the air-to-air combat targets, dropping with increasing angle of attack, corresponding to the selected LSRI-concept. Thrust support in the roll axis was not necessary, thus simplifying the hardware of the nozzle kinematic. The yaw axis ASC-control power showed to be the most critical, because of rapid yawing moment fading with increasing A.O.A. and the resulting speed loss. But with an additional yaw nozzle loop, the aircraft could be held spin free and controlable, reaching the objectives of 0.5 rad/sec² yaw acceleration at high angle of attack.

4. CONCLUDING REMARKS

Summing up, we can make the following conclusions:

The presented configuration with relaxed static stability margin and spin susceptibility at medium angles of attack, could be kept absolutely under control at very fast and tactical orientated control demands due to the selected control laws and feedback signals.

The MIL 8785B specifications for medium n_{z_α}-levels could be attained at PST-conditions, thus not changing aircraft response characteristics at very low speeds, relative to the conventional flight envelope.

Thrust vectoring with conical deflectable engine nozzles showed to be a very powerful moment generator for stabilizing and maneuvering the aircraft up to 70° A.O.A. and resulting speed levels less than 100 knts.

The PST-maneuvers are very dynamically flown and of only short duration time, giving up the high energy status for positional advantage.

With a responsive VNC, both pointing away the aircraft from its flight path and flight path vectoring with different maneuver types showed to be possible. This requires highly responsive actuators in both ASC and VNC-loops with deflection rates of about 50°/sec.

5. REFERENCES

[1] HERBST W.: Future Fighter Technologies
J.A. Vol. 17/8 (Aug. 1980)

[2] HERBST W.: Design for Air Combat 1990
AGARD FMP, Florenz (Oct. 1981)

[3] HERBST W.: Dynamic of Air Combat
24th Israel Annual Conf. on Aviation and Astronautics (Febr. 1982)

[4] KRAUS W.: Delta Canard Configuration at High Angle of Attack
12th ICAS Congress, München (Oct. 1980)

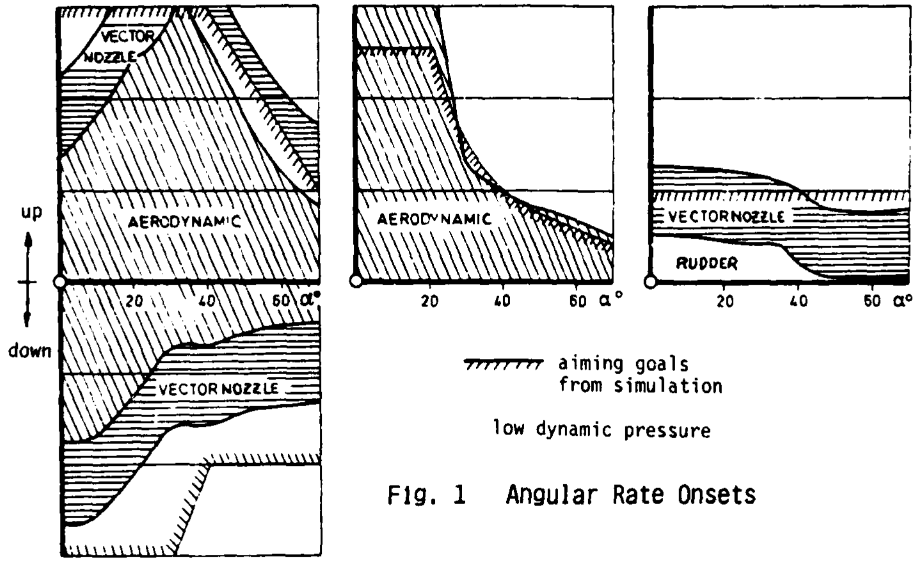


Fig. 1 Angular Rate Onsets

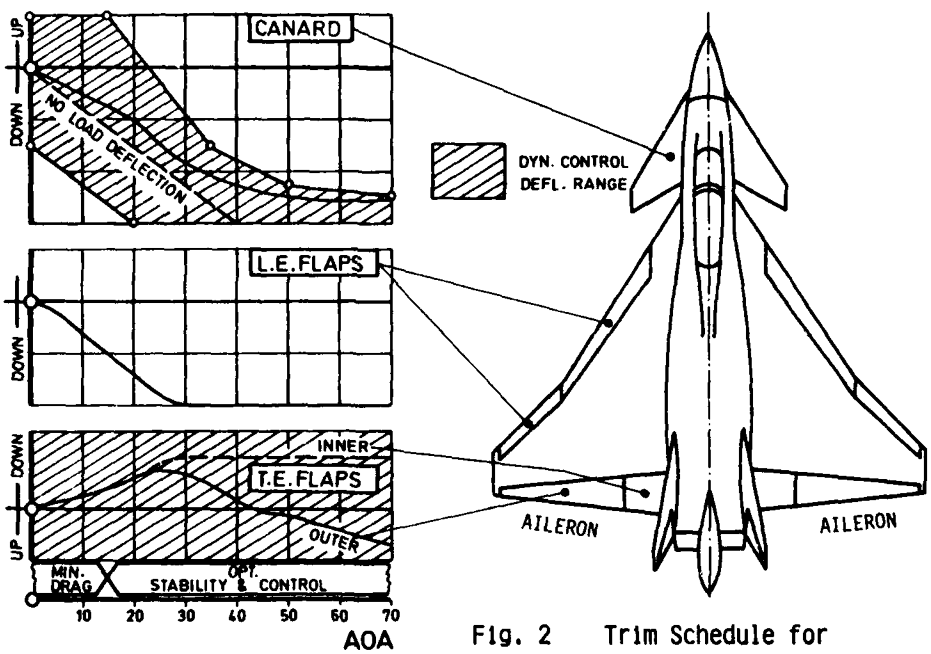


Fig. 2 Trim Schedule for Flaps and Canard

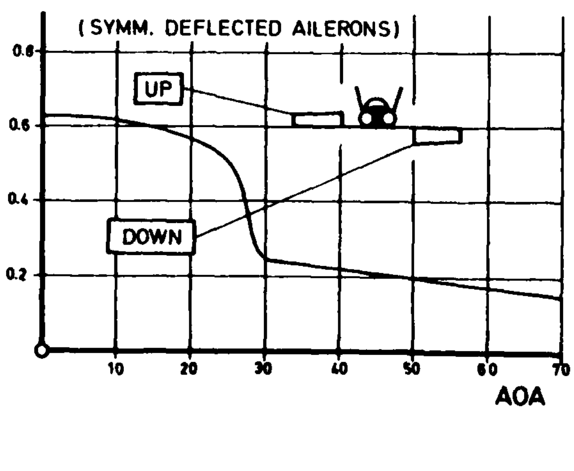


Fig. 3

Maximum Rolling Moment

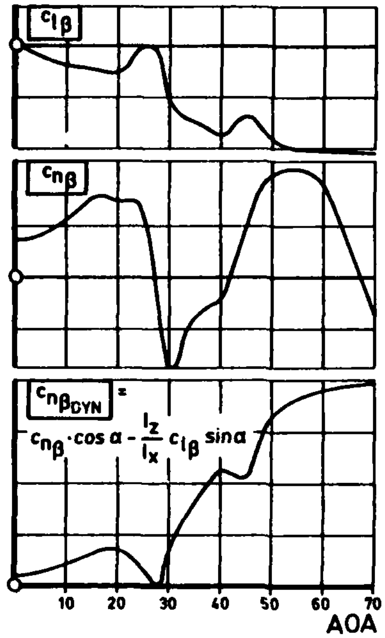


Fig. 4
Lateral and Directional Stability

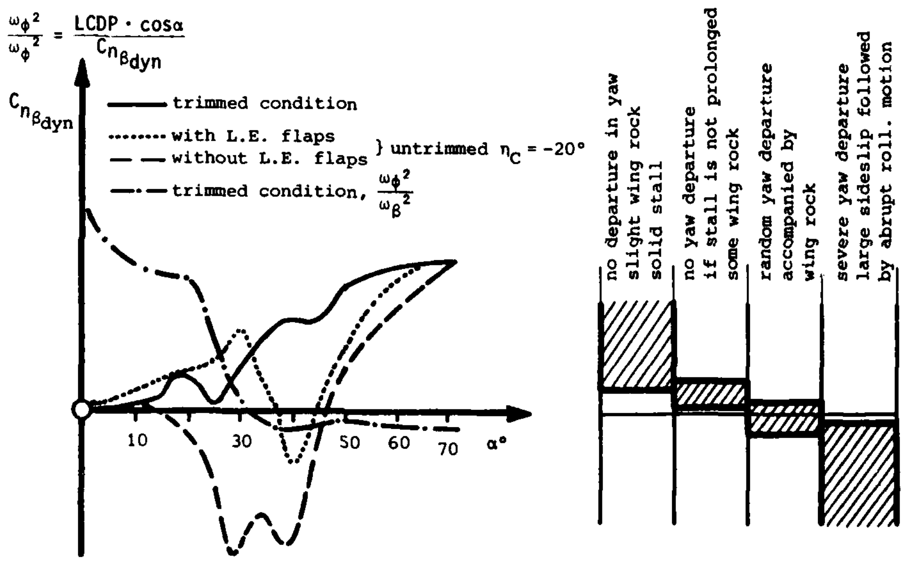


Fig. 5 Departure Parameters

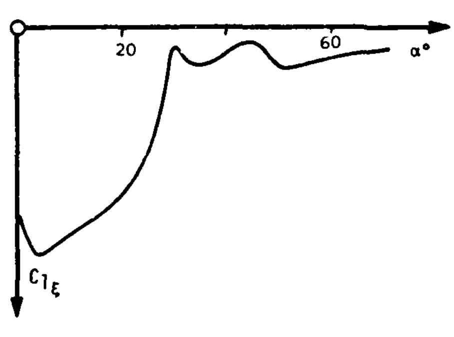


Fig. 6 Aileron Effectiveness

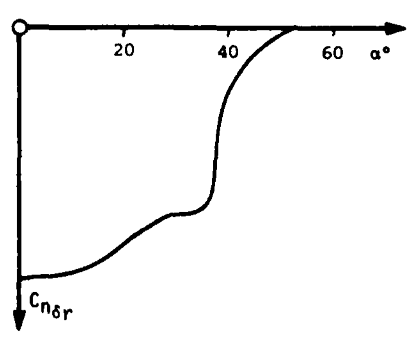


Fig. 7 Rudder Effectiveness

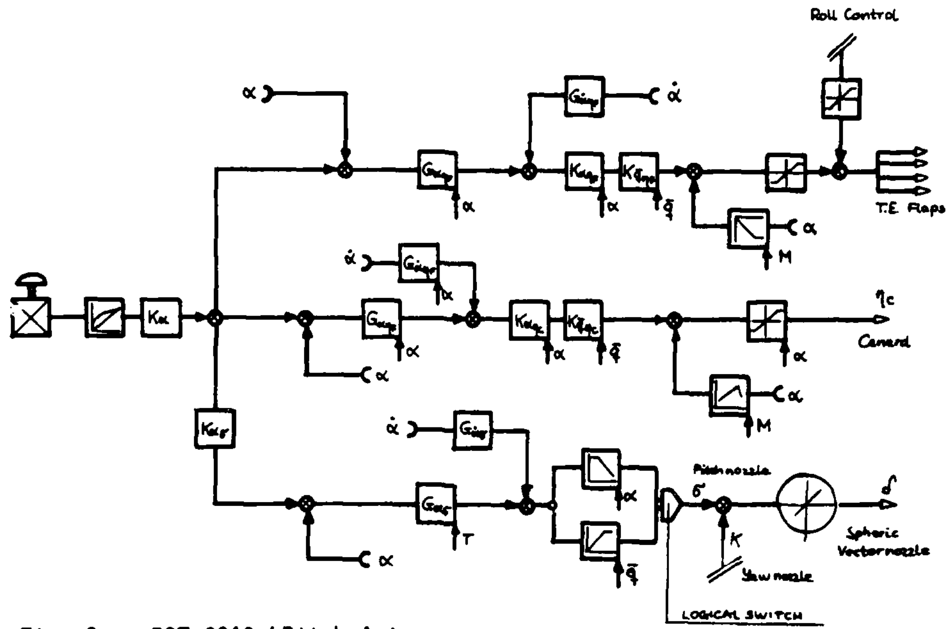
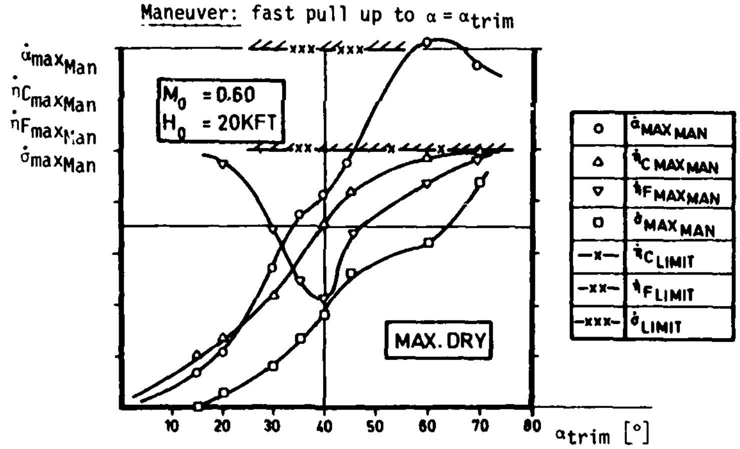


Fig. 8 PST-CSAS / Pitch Axis

Maneuver: fast pull up to $\alpha = \alpha_{trim}$



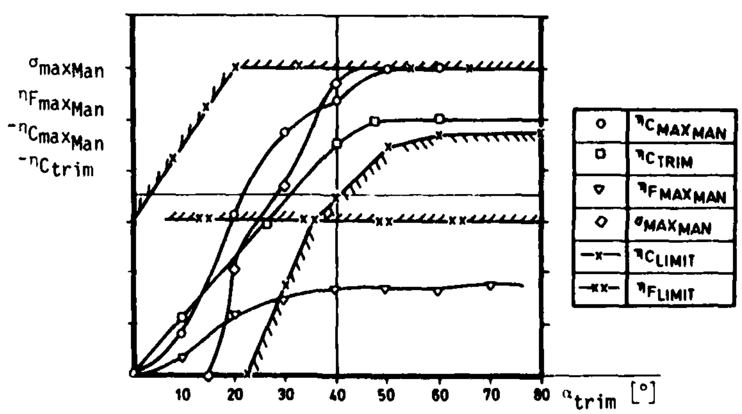


Fig. 9 Simulation Results - pitch axis -

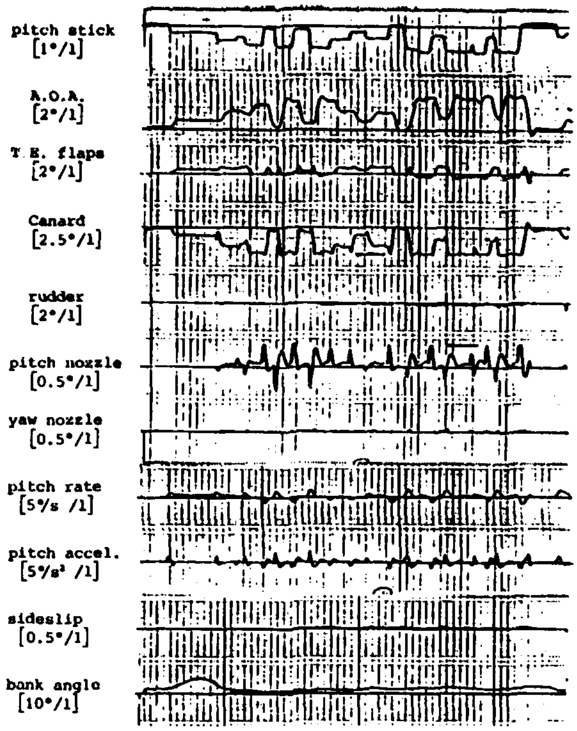


Fig. 10

Simulation Results / Pitch Axis

"Max. Dry", M = 0.6, 20 kft, with VNC

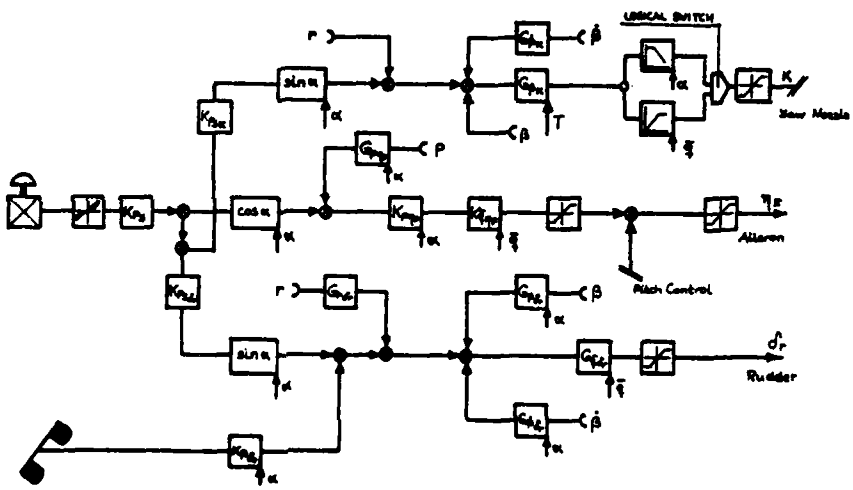


Fig. 11 PST-CSAS / Roll- & Yaw Axis

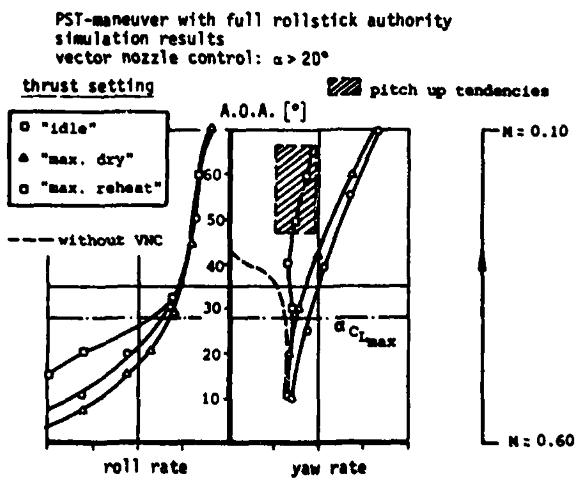


Fig. 12

Rate Performance versus A.O.A. for Three Thrust Settings

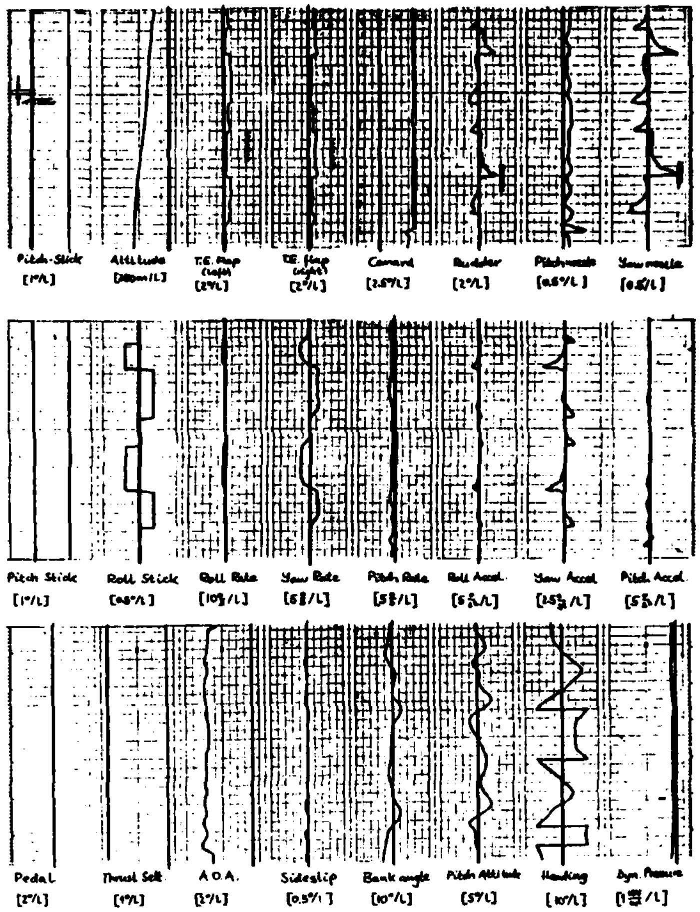


Fig. 13 Simulation Results / Roll- & Yaw Axis
"Max. Reheat", Dyn. Press.: 2KN/m², with VNC

Maneuver type: horizontal turn (180° heading change)
Control status: with VNC (10° nozzle); Initial conditions: $M_0 = 0.6$, $H_0 = 15$ kft, 1g, $\phi_0 = 90°$
Thrust setting: "Max. Dry"

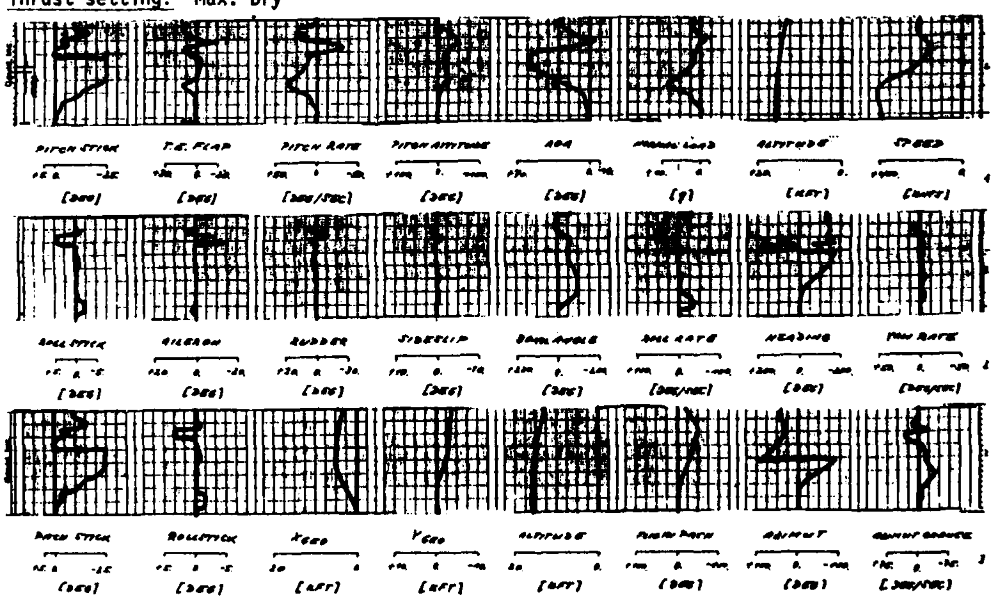


Fig. 14 Simulation Results

Maneuver type: Vertical turn (180° heading change)
Control system: With VNC (10° nozzle); Initial conditions: $M_0 = 0.6$, $H_0 = 15\,kft$, 1g, $\phi_0 = 0°$
Thrust setting: "Max. reheat"

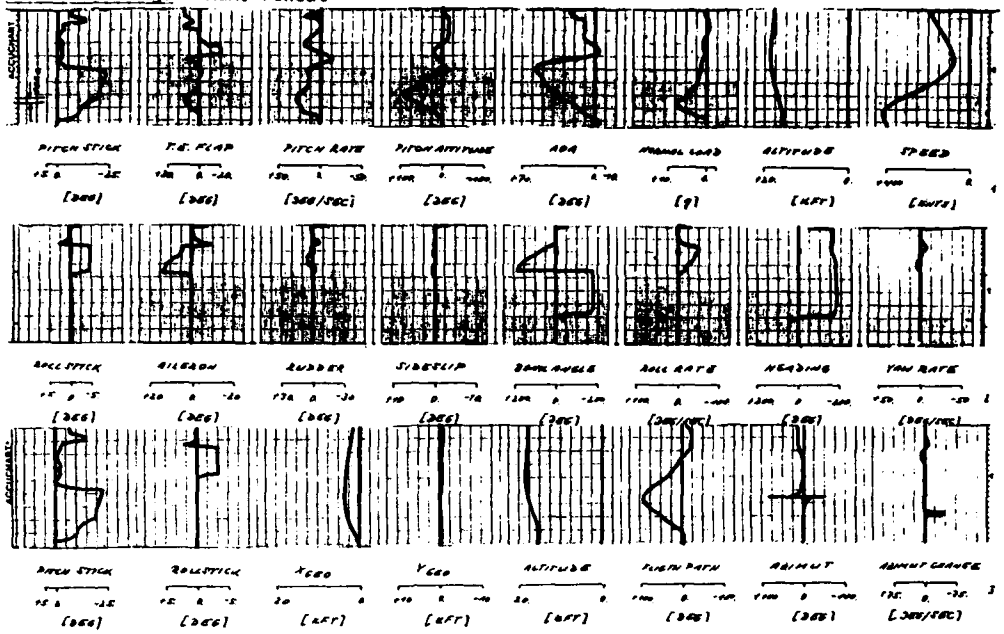
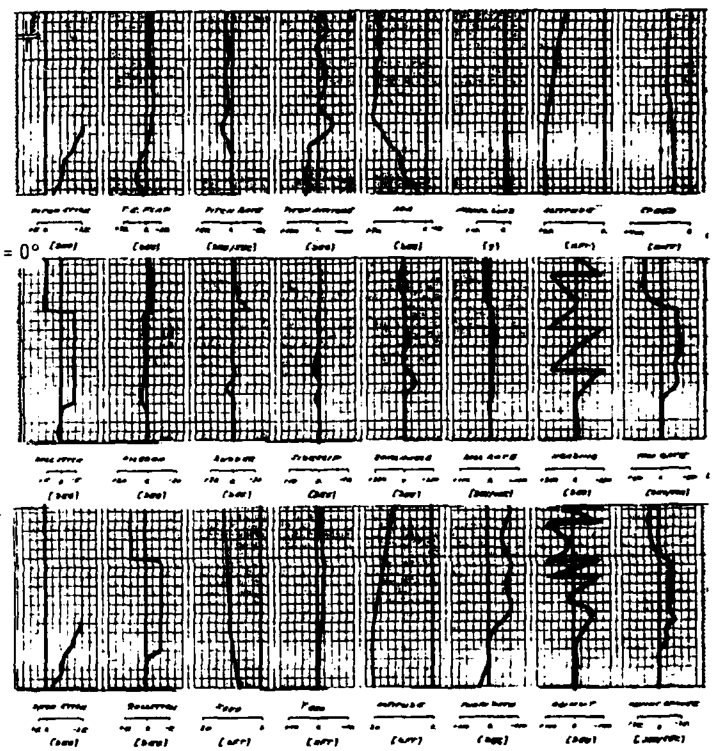


Fig. 15
Simulation Results



Maneuver type: Steady slice
Control status: With VNC (10° nozzle);
Initial cond.: $M_0 = 0.6$, $H_0 = 15\,kft$, 1g, $\phi_0 = 0°$
Thrust setting: "Max. Dry"

Fig. 16
Simulation Results

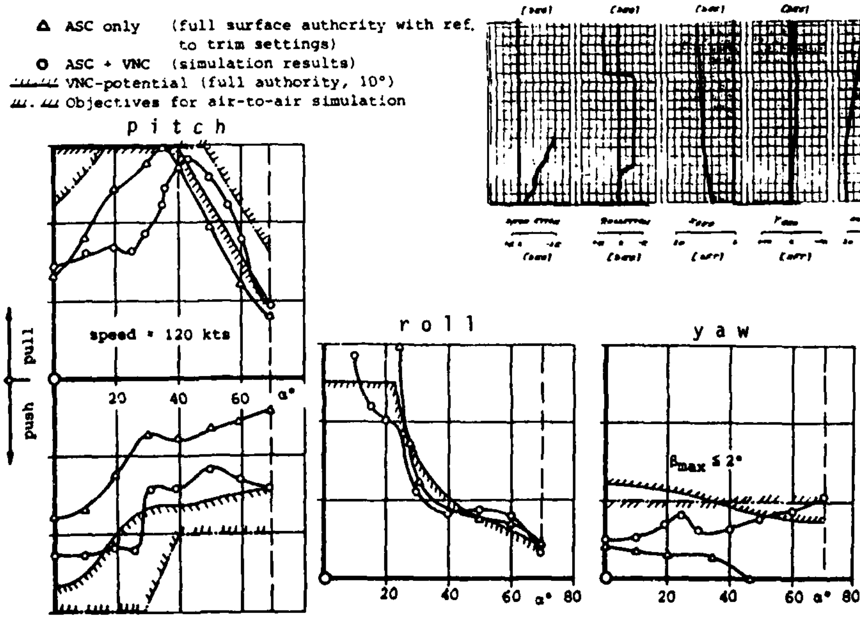


Fig. 17
Angular Rate Onsets
(Body Axis)

EXPERIENCE OF NON-LINEAR HIGH INCIDENCE AERODYNAMIC CHARACTERISTICS

by

D Booker — Principal Aerodynamicist
and
K McKay — Senior Flight Test Engineer
British Aerospace Public Limited Company
Aircraft Group, Warton Division
Warton Aerodrome
Preston, Lancs., PR4 1AX
England

SUMMARY

The impact of aerodynamic non-linearities on aircraft behaviour at high angle of attack is considered from both an analytical and a flight test experience viewpoint. The form of some of these non-linearities and their importance to the design of high incidence control systems is examined. Considerations to be applied to the design of future military aircraft for operation at high angle of attack are suggested.

1. INTRODUCTION

On an aircraft fitted with an advanced high incidence control system the self-restraint normally exercised by pilots when manoeuvring at high angles of attack can be relaxed. This allows freedom to perform inertially coupled, large perturbation roll and pitch manoeuvres which can result in a combination of high sideslip angles and high angles of attack. The presence of aerodynamic non-linearities is to be expected in such conditions.

This paper considers from both analytical and flight experience viewpoints the impact of certain aerodynamic non-linearities on aircraft behaviour at high angle of attack. Effects found to be important when designing aircraft and flight control systems for this flight regime are highlighted. The paper concludes with suggested considerations to be applied to the design of future military aircraft required to operate at high angles of attack.

2. REGIONS WITH NON-LINEAR AERODYNAMICS

2.1 Angles of Attack

Typical handling phenomena experienced as angle of attack is increased as shown in Figure 1. Buffet onset is followed by "random" wing rocking. A progressive deterioration in lateral/directional handling characteristics begins from some higher angle of attack and eventually a departure threshold is passed where spins can occur, either naturally, or possibly when provoked by application of pro-spin controls. Non-linear aerodynamic characteristics are a feature of the region of deteriorating handling qualities and must be considered for any analytical studies of aircraft behaviour in this region.

2.2 Speeds

A typical envelope of allowable angle of attack against speed is shown in Figure 2. For moderate to high speeds the structural nW boundary to which the aircraft is designed keeps down the maximum angle of attack for which an operational clearance is needed. However, at low to moderate speeds routine operation is desired up to (and beyond?) the angle of attack for maximum lift. As described in Section 2.1 the lateral/directional characteristics of the aircraft may intrude on this ideal situation, a handling boundary being defined by, for example, nose-slice, unstable dutch roll, etc (Reference 1). Speeds relevant to lateral/directional misbehaviour, where aerodynamic non-linearities need to be considered, are shown by the cross-hatching on Figure 2.

3. MATHEMATICAL MODELS

For analytical studies of aircraft behaviour at high angle of attack and for designing high incidence flight control systems it is necessary to define a suitable mathematical model. The formulation of such a model is discussed in Reference 2. Non-linear aerodynamics are usually catered for in these models by use of coefficients which are functions of the necessary independent variables. Examples of some of these dependencies are given in the next Section of the paper. Experimental facilities used to define both the linear and non-linear terms in the model include:-

- Wind Tunnel Models
  - Static mounting, in uniform or curved flow (Reference 3)
  - Oscillatory rig mounting (References 3, 4)
  - Rolling rig mounting (Reference 5)
- Free-Flight Models
  - Helicopter or carrier aircraft launched (References 2, 6)
  - Catapult launched (Reference 7)
  - Launched in a vertical wind tunnel (References 8, 9)
  - Flown in a horizontal wind tunnel (Reference 9).

4. TYPES OF NON-LINEARITY

Whilst aerodynamic coefficients can be non-linear in both the longitudinal and lateral/directional senses, discussion in the present paper covers only non-linearities in lateral/directional coefficients. All coefficients are functions of angle of attack and for present purposes this is not considered to be a *non-linearity*. This term is reserved for non-linear dependencies on sideslip angle, roll rate, yaw rate and tail angle. Roll control power and, to a lesser extent, yaw control power can be non-linear but these are not considered here.

Non-linearities due to yaw rate are normally important only for spins. Non-linearities in roll damping occur near wing stall as illustrated in Figure 3. This leads to dependence of dutch roll damping on the amplitude of the oscillations.

The effect of tail angle on the variation with angle of attack of the directional and lateral stability derivatives of a current fighter aircraft is shown in Figure 4. "Stick back" in the context of this figure implies large negative tail angles such as those required to trim at high angle of attack (on an aircraft which is naturally longitudinally stable). Thus trimming significantly reduces $C_{n\beta}$ and $C_{n\beta}$ dynamic at high angle of attack and such effects must be taken into account at the design stage for an aircraft and its flight control system. (It is perhaps worth mentioning that this effect would be less significant on a Relaxed Static Stability aeroplane since tail angle to trim at high angle of attack would be much less negative or possibly positive).

Figure 5 shows the variations of rolling moment and yawing moment with sideslip angle at moderate and high angle of attack. For the lower angle of attack rolling moment is essentially linear and for the configuration shown the yawing moment non-linearity is favourable. At the higher angle of attack the non-linearities in both rolling moment and yawing moment are unfavourable.

The direct effects of large underwing stores on the sideslip characteristics of the configuration are shown in Figure 6. At the lower angle of attack the stores are laterally and directionally destabilising and cause the non-linear variation of yawing moment with sideslip to become unfavourable. At the higher angle of attack the stores have only small effects on the yawing moment characteristic though rolling moments are markedly affected.

However for assessing the overall effects of such stores secondary effects must be considered. With underwing stores the aircraft's longitudinal stability is reduced and thus the tailplane angle to trim at high angle of attack is less negative. The adverse effect of tail angle to trim (Figure 4) is therefore reduced and the net effect of the stores is to slightly improve trimmed directional stability for small sideslip angles; however at high sideslip angles the effects of stores remain adverse.

5. DESIGN FOR CAREFREE MANOEUVRING

The most important non-linearities identified in Section 4 are those associated with high angles of attack and large sideslip angles. Experience at BAe Warton has shown that such conditions can be achieved in flight by performing inertially coupled, large perturbation roll and pitch manoeuvres. Pilots can safely perform such manoeuvres on an aircraft with an advanced high incidence control system which can provide a carefree manoeuvre capability.

To provide carefree manoeuvring in all configurations a simple control system must cope with the aerodynamic characteristics of the most severe case. This could involve some compromise to the manoeuvre capability of less severe configurations. To avoid such compromises either a more complex, and therefore more costly, control system is needed or the carefree manoeuvre capability must be provided only for those configurations for which it is operationally necessary. This topic is worthy of discussion between industry and operators.

6. FLIGHT TEST EXPERIENCE

Figure 7 shows flight test experience of indicated angle of attack (Airstream Direction Detector probe reading) and sideslip angle at departure from controlled flight; the high incidence control system was switched off for these tests. Each symbol represents a separate test. The ADD at departure reduces with increasing sideslip as one would expect with the non-linear aerodynamic characteristics described in Section 4.

An interesting feature is the difference in ADD recorded by the windward and leeward probes (these probes are on the sides of the fuselage just forward of the cockpit area); the windward probe reads much higher ADDs than the leeward probe. This points the way to successful automatic protection against loss of control even when the pilot is carrying out the most severe dynamic manoeuvres. That is to use the higher of the two ADD measurements to provide the incidence signal for the high incidence control system. A control system which uses this approach has been flight tested at BAe Warton and has proved very satisfactory.

7. CONCLUDING REMARKS

Based on experience with BAe Warton projects and with the background information presented in this paper the following considerations are suggested for design of future military aircraft required to operate at high angles of attack.

1. Design for reasonable spin resistance.
2. Ensure controllability at high angle of attack is good.
3. Consider augmenting directional stability to minimise difficulties due to non-linearities with sideslip at high angle of attack. However be aware of other possibly adverse effects, for example the effects of tailplane angle of trim on directional stability.
4. Where a twin incidence probe installation, with probes mounted either side of the front fuselage, is used to provide a monitored incidence signal to the high incidence flight control system the probes are likely to respond differentially to sideslip. The higher reading can be used by the control system to provide additional protection against adverse aerodynamic non-linearities due to sideslip.
5. Decide the role of the aircraft with each store load. Configure the flight control system to provide a carefree manoeuvre capability only for configurations that need it; avoid compromising the manoeuvrability of these configurations by covering more severe but less important ones. Prohibit gross dynamic manoeuvring on those configurations where the operational rôle of the aircraft does not demand it.

8. REFERENCES

1. Lamar, W E: Effects of Buffeting and other Transonic Phenomena: Paper 20B of AGARD-CP-187, 1976.

2. Thomas, H H B M and Edwards, G: Mathematical Models of Aircraft Dynamics for Extreme Flight Conditions (Theory and Experiment): Paper 27 of AGARD-CP-235, 1978.

3. Chambers, J R, Grafton, S B and Lutze, F H: Curved-Flow, Rolling-Flow and Oscillatory Pure-Yawing Wind-Tunnel Test Methods for Determination of Dynamic Stability Derivatives: Paper 7 of AGARD-LS-114, 1981.

4. O'Leary, C O: Wind-Tunnel Measurement of Aerodynamic Derivatives using Flexible-Sting Rigs: Paper 5 of AGARD-LS-114, 1981.

5. Malcolm, G N: Rotary and Magnus Balances: Paper 6 of AGARD-LS-114, 1981.

6. Chambers, J R and Iliff, K W: Estimation of Dynamic Stability Parameters from Drop Model Flight Tests: Paper 11 of AGARD-LS-114, 1981.

7. Vanmansart, M G and Tristrant, R: Etat de l'Art et Perspectives Nouvelles Relatives a l'Etude de la Perte de Contrôle et des Vrilles: Paper 16 of AGARD-CP-319, 1981.

8. Gobeltz, J: Simulation de Vol par Maquettes de Vol Libre en Laboratoires: Paper 14 of AGARD-CP-187, 1976.

9. Chambers, J R, Bowman, J S, Jr and Malcolm, G N: Stall/Spin Test Techniques used by NASA: Paper 13 of AGARD-CP-199, 1976.

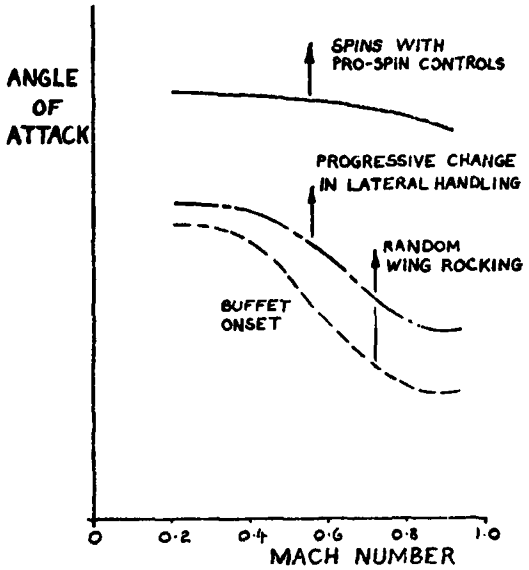


FIG. 1 HANDLING PHENOMENA

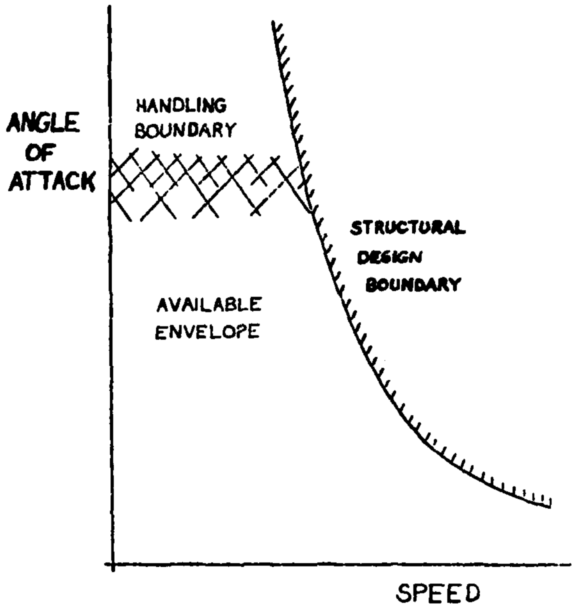


FIG. 2 ALLOWABLE AOA vs SPEED

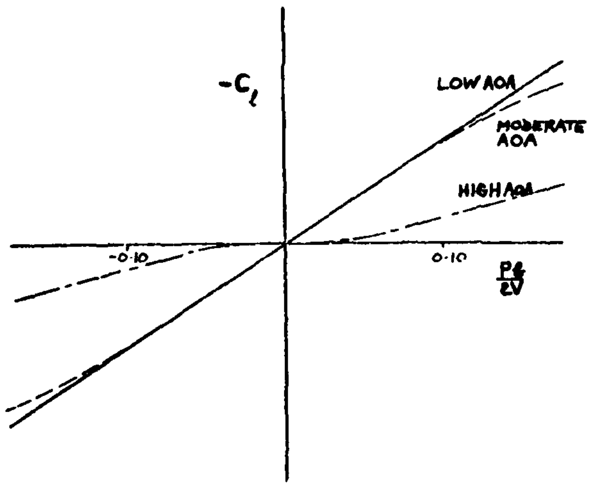


FIG. 3 ROLL DAMPING vs ROLL RATE

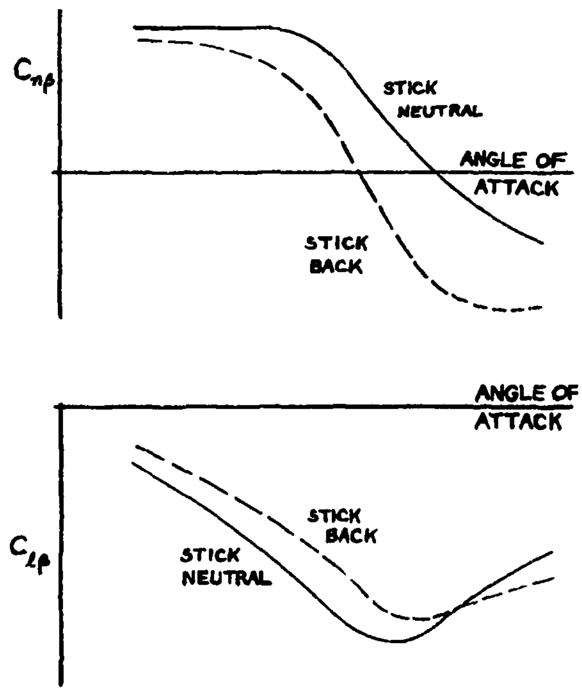


FIG. 4 EFFECT OF TAIL ANGLE ON SIDESLIP CHARACTERISTICS

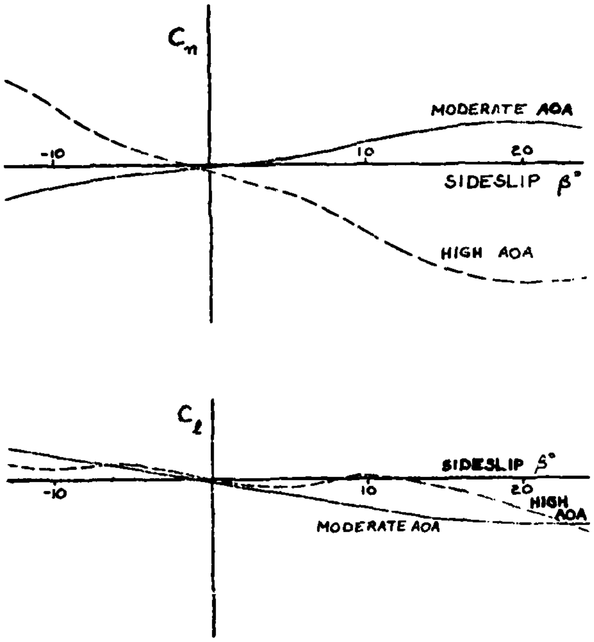


FIG. 5 VARIATION OF YAWING AND ROLLING MOMENTS WITH SIDESLIP

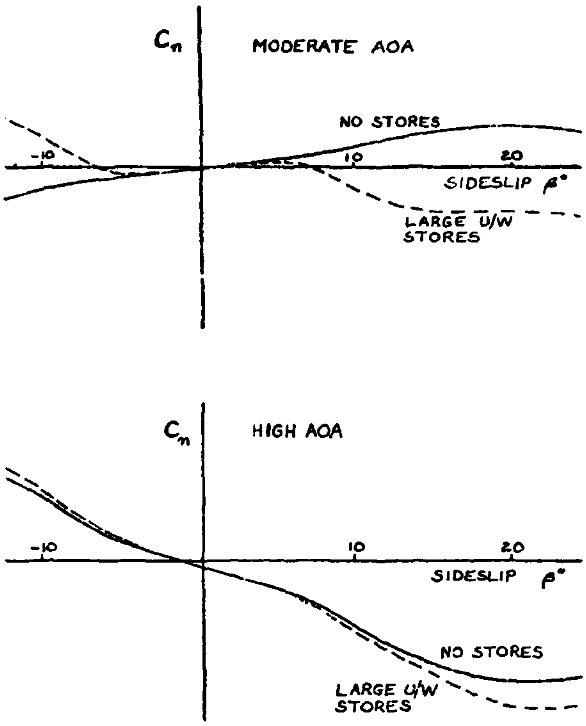


FIG.6a EFFECT OF STORES ON YAWING MOMENT vs SIDESLIP

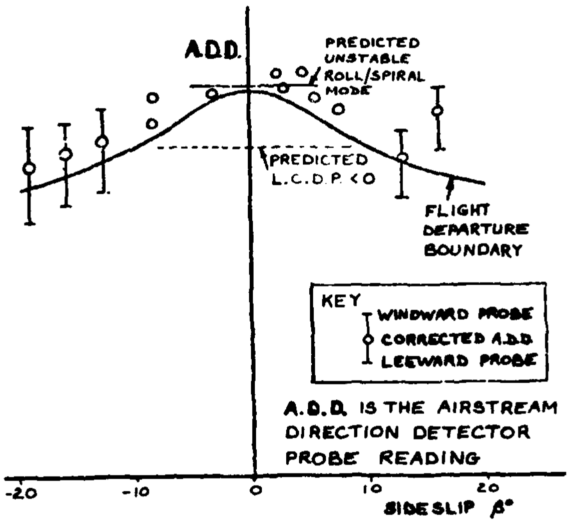


FIG.7 DEPARTURE EXPERIENCE FROM FLIGHT TESTING

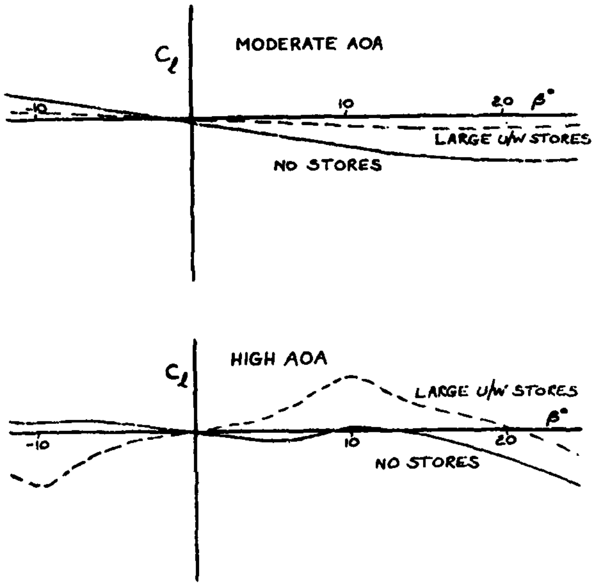


FIG.6b EFFECT OF STORES ON ROLLING MOMENT vs SIDESLIP

A COMPARISON OF ANALYTICAL TECHNIQUES FOR PREDICTING STABILITY BOUNDARIES FOR SOME TYPES OF AERODYNAMIC OR CROSS-COUPLING NONLINEARITIES

by
A. Jean Ross
Aerodynamics Department,
Royal Aircraft Establishment,
Farnborough,
Hants.
U.K.

SUMMARY

The need to predict stability boundaries for flight at high angles of attack is of continuing importance, and the possibility of using analytical techniques, rather than studying computed responses, remains attractive. Two methods of analysis are described and compared, for particular forms of non-linearities, and a relationship is established between nonlinear stability characteristics and linear stability boundaries in terms of the magnitudes of the response variables. The techniques are being used to predict some of the flight characteristics likely to occur for a High Incidence Research Model, which is being tested to provide wind-tunnel and free-flight data for establishing mathematical models of aero-dynamics at high angles of attack. The configuration and the research programme are described briefly.

1 INTRODUCTION

Although it is only possible to specify general descriptive criteria for handling qualities for departure and spinning characteristics of aircraft, it is still necessary to quantify the characteristics in some way for assessment purposes during the development of an aircraft. The first problem is to specify the form of the mathematical model of the aerodynamic forces and moments, and then to specify the numerical values of the parameters needed to reproduce the responses to control inputs at high angles of attack. There have been a number of research papers published recently describing either the mathematical model of particular aircraft (eg Ref 1), or describing the types of nonlinearity which may have to be included for adequate representation of the aerodynamics of current and future aircraft, for flight at high angles of attack[2,3]. The next problem is to obtain an overall picture[4,5] of the characteristics of the dynamic responses to all types of control inputs likely to be used, since, without a method of analysis to define the characteristics, the number of response calculations (or simulations) required is very large. It is not possible to guarantee that all possible types of motion have been encountered in these calculations, due to the dependence of nonlinear systems on the amplitudes and phases of the response variables. A familiar example of this is the uncertainty as to whether all the spin modes of a new aircraft have been encountered in either simulations, or tests in a spin tunnel, or in free-flight tests of models, before the actual aircraft is flown to high angles of attack. Another important example is the lengthy and possibly inconclusive process needed to assess and test control systems which are designed to prevent departure from controlled flight.

Some progress has been made recently in obtaining general stability characteristics for various particular types of nonlinearity likely to be present in the kinematic and aerodynamic terms in the equations of motion of aircraft, and two methods of analysis[5,6] are discussed in section 2 (Description of methods) and section 3 (Examples) of this paper. The preliminary analysis needed for determining the possible non-zero equilibrium states of the nonlinear system is also described, the emphasis being on defining the complexity of nonlinear terms which can be included for algebraic solutions to exist. Such solutions can give insight to the significance of various terms much more readily than those which have to be obtained using an iterative numerical process.

The first method of analysis involves the familiar linearisation of the equations of motion about the non-zero equilibrium state, which has also been described in Ref 5 in terms of Catastrophe Theory and Bifurcation Analysis. The results in Ref 5 are mainly presented in terms of the magnitudes of steady control inputs which lead to zero damping of either exponential or oscillatory modes (bifurcation surfaces), but it is shown here that for many responses, the magnitude of one of the response variables is the basic significant parameter. In particular, the well-known significance of roll-rate in inertia cross-coupling is demonstrated, using Bifurcation Analysis.

The second method of analysis is a simple extension[7] of the averaging technique introduced by Krylov, Bogoliuboff and Mitropolsky, to obtain approximate analytic solutions for the nonlinear "damping" and "frequency" of nonlinear "modes" in terms of the amplitude of response. A relationship between the linear stability characteristics given by Bifurcation Analysis and the nonlinear stability characteristics is obtained for a particular type of nonlinearity in aerodynamic moments due to sideslip, which again demonstrates that the amplitude of the response variables is the significant parameter in defining the type of nonlinear response which could occur, such as limit cycle, oscillatory divergence or exponential-type divergence. The possibility of introducing other forms of nonlinearity is also discussed briefly.

These methods are being applied to a mathematical model of a particular configuration, the High Incidence Research Model, which is the subject of a theoretical and experimental research programme[8] aimed at establishing techniques for mathematical modelling at high angles of attack. Static and dynamic aerodynamic data are currently being obtained from wind-tunnel tests, and flight data will be obtained using free-flight models. The configuration and research programme are described in Section 4, as they are the current focus for the theoretical work described earlier.

2 DESCRIPTION OF THEORETICAL TECHNIQUES

2.1 Equilibrium States for Nonlinear Equations of Motion

2.1.1 Equations of motion and datum state

The search for meaningful approximate solutions to the nonlinear equations for the equilibrium states of an aircraft seems to have been revived recently, particularly for analysis of steady spins[9,10]. Digital computer techniques are now available[11], but there is still need for algebraic solutions - even if approximate - so that important parameters for new aircraft shapes may be identified. Associated with such searches are the various suggestions for appropriate state variables, ie which quantities to use to express the mathematics. An interesting collection[12] relevant to spin has recently been published, but there does not seem to be an ideal set of variables. Two distinct sets of angles are used, Euler angles defining the orientation of the aircraft relative to earth, and the incidence angles defining the orientation relative to the velocity vector. These sets are related via the flight path angles, and so the equations are necessarily complicated. For the angular motion, it is possible to use either the familiar components about the moving body axes, or to use the spin rate at given radius about a given axis relative to the earth. It is not proposed to discuss these representations more fully here, but to demonstrate which types of nonlinearity can be included to still yield equilibrium equations which have analytic (algebraic) solutions.

The general equations of motion of an aircraft may be written as:

$$\dot{u} = rv - qw + \frac{X}{m} - g \sin \Theta \tag{1}$$

$$\dot{v} = pw - ru + \frac{Y}{m} + g \sin \Phi \cos \Theta \tag{2}$$

$$\dot{w} = qu - pv + \frac{Z}{m} + g \cos \Phi \cos \Theta \tag{3}$$

$$\dot{p} = \frac{I_{xz}(\dot{r}+pq)}{I_x} + \frac{(I_y-I_z)qr}{I_x} + \frac{L}{I_x} \tag{4}$$

$$\dot{q} = \frac{I_{xz}(r^2-p^2)}{I_y} + \frac{(I_z-I_x)pr}{I_y} + \frac{M}{I_y} \tag{5}$$

$$\dot{r} = \frac{I_{xz}(\dot{p}-qr)}{I_z} + \frac{(I_x-I_y)pq}{I_z} + \frac{N}{I_z} \tag{6}$$

For the determination of developed equilibrium states in which $\dot{u}=\dot{v}=\dot{w}=\dot{p}=\dot{q}=\dot{r}=0$, such as autorotational rolling or spinning, it is usually justifiable to assume that the gravity terms may be accounted for as an averaged contribution, ie a symmetric steady state is assumed to exist, at which the normal force due to angle of attack is balanced by the mean value of ($-mg \cos \Theta$), and the associated axial force is balanced by the mean value of ($mg \sin \Theta$). The residual oscillatory contribution from the gravity terms is of small amplitude, and can usually be neglected. The pitching moment has also to be balanced by an elevator deflection, and so the symmetric datum state may be described in terms of the variables u_o, w_o, η_o and resultant velocity V_o. The aerodynamic forces and moments may also be expressed in terms of this datum state, and in particular the datum angle of attack, $\tan^{-1}(w_o/u_o)$.

2.1.2 Equilibrium states for linear aerodynamic terms

The nonlinearities in the kinematics define additional non-zero equilibrium states, denoted here by suffix 'e', and it is useful to define w_e, η_e as the increments, not necessarily small, to the datum state, so that $w = w_o+w_e$, $\eta = \eta_o+\eta_e$, $Z = Z_o+Z_e$ etc, but where $\eta_e = 0$ does not imply that $w_e = 0$. However, the resultant velocity remains near-constant, and so the assumption is made that $u = u_o$, to make the analysis tractable. It is also useful to choose the principal axes of inertia as the reference axes, so that $I_{xz} = 0$, simplifying the form of equations (4), (5) and (6) without loss of accuracy. If the aerodynamic forces and moments are initially assumed to be linearly dependent on the response variables, and are expressed in terms of the familiar stability derivatives at the datum angle of attack, (but including all the cross-coupled derivatives such as l_q), the equilibrium equations for the variables $\hat{v}_e = \frac{v_e}{V}$, $\hat{w}_e = \frac{w_e}{V}$, p_e, q_e, r_e may be written as:

$$\begin{bmatrix} y_v & p_e+y_w & \hat{w}_o+y_p & y_q & -\hat{u}_o+y_r \\ -p_e+z_v & z_w & z_p & \hat{u}_o+z_q & z_r \\ l_v & l_w & l_p & l_q & l_r+b_x q_e \\ m_v & m_w & m_p & m_q & b_y p_e+m_r \\ n_v & n_w & n_p & b_z p_e+n_q & n_r \end{bmatrix} \begin{bmatrix} \hat{v}_e \\ \hat{w}_e \\ p_e \\ q_e \\ r_e \end{bmatrix} = - \begin{bmatrix} y_c \\ z_c \\ l_c \\ m_c \\ n_c \end{bmatrix} \tag{7}$$

where the concise dimensional form of the derivatives has been used (ie the dimensional aerodynamic derivative divided by the appropriate mass or moment of inertia), $b_x = (I_y-I_z)/I_x$ etc, and suffix c on the RHS denotes force or moment due to control deflections. The variables $\hat{v}_e$ and $\hat{w}_e$ have been used, rather than α and β, since the complete definitions of the angles would be needed, eg $\hat{w}_e = \sin \alpha_e$.

These equations may be solved using determinants, by observing that the determinant of the chosen form* of the matrix in equations (7) is linearly dependent on q_e, and is a polynomial of fourth order in p_e. Thus it is possible to solve for q_e and p_e, in particular, to give expressions of the form:

$$p_e = \frac{P_o(p_e) + P_1(p_e)\, q_e}{D_o(p_e) + D_1(p_e)\, q_e} \tag{8}$$

$$q_e = \frac{Q_o(p_e) + Q_1(p_e)\, q_e}{D_o(p_e) + D_1(p_e)\, q_e} \tag{9}$$

Equation (8) gives the solution for q_e in terms of p_e, which may be substituted into equation (9), to yield a polynomial in p_e.

$$\begin{aligned}&\left[P_o(p_e) - p_e D_o(p_e)\right]\left[D_1(p_e)P_o(p_e) - D_o(p_e)P_1(p_e)\right] \\ &+ \left[p_e D_1(p_e) - P_1(p_e)\right]\left[Q_o(p_e)\{P_1(p_e) - p_e D_1(p_e)\} - Q_1(p_e)\{P_o(p_e) - p_e D_o(p_e)\}\right] = 0\end{aligned} \tag{10}$$

In general, the polynomials D_o and P_o are quartic in p_e, D_1 P_1 and Q_o are cubics, and Q_1 is a quadratic, so that the polynomial in equation (10) is of order 12. There is some simplification if the cross-coupling derivatives are neglected, but the form and order of equation (10) remains the same. If some of the damping terms are neglected, such as m_q,[10] then the order of the polynomial is reduced to 7, and if all damping terms (derivatives due to p, q and r) are neglected, then a quartic is obtained.

For the special case of inertia cross-coupling, where the equilibrium state is independent of control setting, then the determinant of the matrix in equation (7) is zero, ie

$$D_o(p_e) + D_1(p_e)q_e = 0 \tag{11}$$

With the assumption that $b_x \approx 0$ (or that the product $b_x q_e r_e$ is negligible), equation (11) gives the equilibrium rates of roll at which auto-rotation can occur, and reduces to the classic bi-quadratic if the stiffness terms and damping-in-roll derivative only are retained. If $b_x \neq 0$, then equation (11) has to be combined with the ratio of p_e/q_e, obtained from equations (8) and (9). With this form of equations, it is possible to eliminate q_e, to obtain a polynomial of order 11 in p_e.

2.1.3 Equilibrium states for nonlinear aerodynamic terms

Having established a method of solution with linear aerodynamics, it is interesting to see how many, and which types of aerodynamic nonlinearity may be introduced before it is impossible to obtain a polynomial in p_e alone. For example Mehra[5] has included a linear dependence of some of the lateral derivatives on angle of attack, and his equations for the equilibrium state may be written in the form:

$$\begin{bmatrix} y_v & p_e & \sin\alpha_o & 0 & -\cos\alpha_o \\ -p_e & z_w & 0 & \cos\alpha_o & 0 \\ (l_v + l_{vw}\hat{w}_e) & l_{\xi w}\xi_c & l_p & l_q & (l_r + l_{rw}\hat{w}_e + b_x q_e) \\ 0 & m_w & 0 & m_q & b_y p_e \\ n_v & (n_{\xi w}\xi_c + n_{pw}p_e) & n_p & b_z p_e & n_r \end{bmatrix} \begin{bmatrix} \hat{v}_e \\ \hat{w}_e \\ p_e \\ q_e \\ r_e \end{bmatrix} = \begin{bmatrix} y_c \\ z_c \\ l_c \\ m_c \\ n_c \end{bmatrix} \tag{12}$$

The determinants are again linear functions of q_e, and also linear functions of $\hat{w}_e$, so it is still possible to obtain a polynomial in p_e, from the equations for p_e, q_e and $\hat{w}_e$, although the algebra is lengthy. Elimination of q_e leads to two simultaneous polynomial equations in $\hat{w}_e$ and p_e, with quadratic terms in $\hat{w}_e$ which may be eliminated, to give a high order polynomial in p_e. It may be observed that it is possible to include the additional derivatives l_{pw} and l_{qw}, (since they do not introduce higher-order terms in $\hat{w}_e$) and still obtain a polynomial in p_e, but that addition of the variation of yawing moment due to sideslip with angle of attack, n_{vw}, leads to a quartic equation in $\hat{w}_e$. In general, the aerodynamic data available indicate dependence of the derivatives on angle of attack primarily, and on angle of sideslip to a lesser extent, with indications from tests on rotary rigs of nonlinear moments due to rate of roll. The latter may easily be accommodated in the determinants above, without causing high order terms in $\hat{w}_e$ or q_e. Pinsker[10] has observed that the normal force can often be expressed as a near-linear function, using $\frac{C_Z}{\cos\alpha_t} = k_o + k_1 \tan\alpha_t$, where $\tan\alpha_t = w/_u$, and so it may be feasible to extend the linear dependence by using this combination. For large angles of attack, pitching moment is also often approximately linear with angle of attack, and so it is the variation of m_q and the lateral derivatives which could be important. Of these, the second-order derivatives l_{vw}, l_{pw}, l_{rw}, $l_{\xi w}$, l_{qw}, n_{pw}, n_{rw}, $n_{\xi w}$ and n_{qw} may be included, to give a polynomial in p_e, but the addition of either of the derivatives n_{vw} or m_{qw} leads to high-order equations in $\hat{w}_e$. It may be noted that any dependence of l_p and n_p on angle of sideslip may also be included.

---

*There is a choice for the second order terms, eg the term $b_y p_e r_e$ may be considered as $(b_y p_e).r_e$ or $(b_y r_e).p_e$.

Several investigators of inertia cross-coupling problems have neglected the '$b_x qr$' term in the rolling moment equation, since $b_x \approx 0$ for aircraft with moderate I_x/I_z ratios, and q_e, r_e much lower than p_e. For such cases, it would be possible to obtain a solution when several second-order derivatives with respect to angle of attack are included in the mathematical model.

For spinning problems, it is usually necessary to retain the '$b_x qr$' term, especially for inertially-slender aircraft, and so the algebraic solution, if possible, is lengthy. The alternative 'classic' method of solution (eg Ref 13), for the spin states is to consider the moment equations, expressing p, q, r in terms of the spin rate and the attitude angles, so that a set of equations are obtained which have to be solved graphically, or iteratively.

Some simpler dynamic problems are also of interest, in particular the conditions for lateral departure from controlled flight, when the mean angle of attack can be considered to be approximately constant for the initial motion. It is possible to include some higher-order aerodynamic nonlinearities, and the solution for the equilibrium states remains straightforward.

2.2 Linear Stability Characteristics

The obvious technique to use for the study of the stability of responses about a non-zero equilibrium state, such as a rapid roll or spin, is to consider only small perturbations about that equilibrium state, so that the equations of motion may be linearised, and the usual stability roots derived. Although the resulting equations [14,15,16] are often of high order, it is usually possible to identify the important parameters, and so gain an insight to the basic characteristics of the responses. For example, the simplest approximation to the response in inertia cross-coupled motion indicates that there are two possible non-zero equilibrium states, the lower of which is unstable, so that the response diverges to the higher stable equilibrium state. This property leads to the concept of a "critical roll rate", above which the response will diverge to the equilibrium state, irrespective of control inputs.

In general, nonlinear equations of motion may be written in the form:

$$\dot{\underline{x}} = f(\underline{x}, \underline{c}), \tag{13}$$

where $\underline{x}$ is an n-dimensional vector of state variables, and $\underline{c}$ is an m-dimensional vector of control variables. The basic equilibrium state is $\underline{x} = 0$, $\underline{c} = 0$ (to give $\dot{\underline{x}} = 0$), but for nonlinear systems, other non-zero equilibrium states may exist, for which

$$f(\underline{x}_e, \underline{c}_e) = 0. \tag{14}$$

The stability characteristics about this equilibrium state may be obtained by considering the incremental vector $\underline{x}'$, where

$$\begin{aligned} \dot{\underline{x}}' &= f(\underline{x}_e + \underline{x}', c_e) \\ &= f(x_e, c_e) + \left[\frac{\delta f}{\delta x}\right]_e . \underline{x}' + \ldots \\ &= F(\underline{x}_e, \underline{c}_e) . \underline{x}' + \ldots \end{aligned} \tag{15}$$

The form of $F(\underline{x}, \underline{c})$ is evaluated by inspection of the form of $f(\underline{x}, \underline{c})$. It is usually possible to write $f(\underline{x}, \underline{c})$ in matrix notation, as in Section 2.1, and then $F(\underline{x}, \underline{c})$ is the Jacobian matrix of $f(\underline{x}, \underline{c})$. For the linearised system, the stability polynomial is obtained by assuming that $\underline{x}' = \underline{A}_x . e^{\mu t}$, to give the determinantal equation,

$$\left| F(\underline{x}_e, \underline{c}_e) - \mu I \right| = 0, \tag{16}$$

where μ may be real or complex, corresponding to exponential or oscillatory modes respectively.

The condition for zero damping of an exponential mode is that $\left| F(\underline{x}_e, \underline{c}_e) \right| = 0$, (17)

so that equations (14) and (17) give n+1 equations to determine the particular n equilibrium states and the relationship between the control settings for which zero damping occurs. Thus the stability boundaries are defined in the control-space, beyond which divergence occurs, and are known as bifurcation surfaces.

The corresponding condition for zero damping of an oscillatory mode is that Real part of [illegible] but this cannot be expressed simply, being the Routh's discriminant of the stability polynomial. However, the boundaries can be computed for given forms of $f(\underline{x}, \underline{c})$, beyond which the oscillatory modes have increasing amplitude, and these are termed Hopf bifurcation surfaces.

The significance of these stability boundaries has been discussed by Mehra [illegible] Analysis and Catastrophe Theory Methodology (BACTM). The nominal equilibrium states [illegible] exponential mode has zero damping are not achievable, since the response either [illegible] occurs) or changes to a stable equilibrium state (ie a jump occurs). When [illegible] equilibrium state has zero damping, then the response may become [illegible] oscillatory manner.

The form of the function $F(\underline{x}, \underline{c})$ may be derived [illegible] second-order nonlinearities of polynomial form, since [illegible] matrix form

$$\dot{\underline{x}} = A \ldots \text{[illegible]}$$

AD-A118 596 ADVISORY GROUP FOR AEROSPACE RESEARCH AND DEVELOPMENT--ETC F/G 1/3
CRITERIA FOR HANDLING QUALITIES OF MILITARY AIRCRAFT.(U)
JUN 82
UNCLASSIFIED AGARD-CP-333 NL

3 of 4

where $A(\underline{x})$ and $B(\underline{x})$ contain only first order terms in $\underline{x}$. An example of $A(\underline{x})$ is given in equation (7), where the matrix on the left hand side of the equilibrium equation contains linear terms in p and q. Thus $A(\underline{x})$ may be written as the sum of two matrices, one of which is independent of the state variables, and the other containing only linear terms. There is a choice in the ordering of this second matrix, since a term in the equations $a_{ij}x_ix_j$ can be considered as either $(a_{ij}x_i)$ in the j'th column to be multiplied by x_j, or as $(a_{ij}x_j)$ in the i'th column, to be multiplied by x_i. One obvious choice is to keep the number of different x_i appearing in the matrix to a minimum, as has been done in equation (7). Thus equation (18) may be expanded to give

$$\begin{aligned}\dot{\underline{x}} &= [A_o+A_1(\underline{x})].\underline{x} + [B_o+B_1(\underline{x})].\underline{c} \\ &= [A_o+A_1(\underline{x}_e) + A_1(\underline{x}')].(\underline{x}_e+\underline{x}') + [B_o+B_1(\underline{x}_e) + B_1(\underline{x}')].(\underline{c}_e+\underline{c}') \\ &= [A_o+A_1(\underline{x}_e)].\underline{x}' + A_1(\underline{x}').\underline{x}_e + [B_o+B_1(\underline{x}_e)].\underline{c}' + B_1(\underline{x}').\underline{c}_e + O(\underline{x}'^2) \\ &= [A_o+A_1(\underline{x}_e) + A_2(\underline{x}_e) + B_2(c_e)].\underline{x}' + [B_o+B_1(\underline{x}_e)].\underline{c}' \end{aligned} \tag{19}$$

where
$$A_1(\underline{x}').\underline{x}_e \equiv A_2(\underline{x}_e).\underline{x}' \tag{20}$$

$$B_1(\underline{x}').\underline{c}_e \equiv B_2(\underline{x}_e).\underline{x}', \tag{21}$$

and the equilibrium condition is

$$[A_o+A_1(\underline{x}_e)].\underline{x}_e + [B_o+B_1(\underline{x}_e)].\underline{c}_e = 0 \tag{22}$$

The stability polynomial is then given by the determinantal equation corresponding to equation (16), viz

$$\left|A_o+A_1(\underline{x}_e) + A_2(\underline{x}_e) + B_2(\underline{c}_e) - \mu I\right| = 0 \tag{23}$$

The bifurcation surfaces are defined by the roots of the stability polynomial for which $R_e(\mu) = 0$, and may be denoted by the parametric relationships between x_b and c_b, ie particular values of x_e and c_e.

An alternative basis for examining stability about non-zero equilibrium states is to choose a particular state, and use equation (22) to determine the control inputs required, for substitution into equation (23), ie the stability determinant may be expressed in terms of the equilibrium state. In particular, if the control characteristics are independent of the state variables (eg control derivatives independent of angle of attack etc) then $B_1(\underline{x}) \equiv 0$, $B_2(\underline{c}_e) \equiv 0$, and the matrix in equation (23) is not explicitly dependent on control inputs. Thus the stability boundaries are explicitly dependent on the state variables only. Thus the concepts of critical magnitudes of certain state variables leading to divergence, such as rate of roll for inertia cross-coupling, and possibly rate of yaw for some spin entries, is probably of wider application to other nonlinear responses, and is being explored further.

2.3 Nonlinear Stability Characteristics

Several techniques are being developed to study the characteristics of the responses of nonlinear dynamic systems, two of the most widely used being Lyapunov functions and the averaging method of Krylov, Bogoliuboff and Mitropolsky. The latter has been extended[7] to include the concept of nonlinear damping, as a function of amplitude, and to high-order differential equations[17], to give approximate analytic solutions. Most of the work at the RAE has been centred on the lateral equations of motion with nonlinear moments due to sideslip or roll rate, to study lateral departures, so that the equations reduce to one differential equation in one variable. The method is described here for an example[6] with linear and cubic terms in the representation of rolling and yawing moments due to sideslip. The equations are given by:

$$\dot{\hat{v}} = y_{v1}\hat{v} + y_{v3}\hat{v}^3 + \hat{w}_o p - \hat{u}_o r + \frac{g}{V}\sin\Phi\cos\Theta_o + y_e \tag{26}$$

$$\dot{p} = l_{v1}\hat{v} + l_{v3}\hat{v}^3 + l_p p + l_r r + l_e \tag{27}$$

$$\dot{r} = n_{v1}\hat{v} + n_{v3}\hat{v}^3 + n_p p + n_r r + n_e \tag{28}$$

where the concise moment derivatives contain product of inertia terms, and contributions from a simple roll-damper with gearing $K_{\xi p}$. The gravity term may be retained for study of responses about the zero equilibrium state, by linearising $\sin\Phi$, but for responses about a non-zero equilibrium state the gravity term has to be neglected. Elimination of p and r then yields one equation in $\hat{v}$, of the form:

$$G(\hat{v}) \equiv \frac{d^3\hat{v}}{dt^3} + \frac{d^2A(\hat{v})}{dt^2} + \frac{d}{dt}B(\hat{v}) + C(\hat{v}) - E(y_e, l_e, n_e) = 0$$

where $A(\hat{v})$, $B(\hat{v})$, $C(\hat{v})$ contain linear and cubic terms in $\hat{v}$, eg $C(\hat{v}) = c_1\hat{v}+c_3\hat{v}^3$. A solution of the form $\hat{v} = \hat{v}_e + \sigma\cos\phi$ is sought, where $\hat{v}_e$ is the equilibrium state, $C(\hat{v}_e) = E(y_e, l_e, n_e)$, and the nonlinear damping and frequency are defined by

$$\frac{d\sigma}{dt} = k\sigma \ , \ \frac{d\phi}{dt} = \omega \tag{29}$$

If the assumption is made that k and ω are approximately constant for one cycle of ϕ, (from 0 to 2π), then the averaging technique gives an approximate solution, for k and ω as functions of σ,

$$\frac{1}{2\pi}\int_o^{2\pi} G.\cos\phi.\,d\phi = 0 \text{ and } \frac{1}{2\pi}\int_o^{2\pi} G.\sin\phi\,d\phi = 0 \tag{30}$$

These two equations may be combined and written in complex form, by considering $\frac{1}{2\pi}\int_0^{2\pi} G[\cos\phi - i\sin\phi]\,d\phi = 0$,

which yields the nonlinear stability equation,

$$(k+i\omega)^3 + (a_1+3a_3\hat{v}_e^{\,2})(k+i\omega)^2 + \tfrac{3}{4}a_3\sigma^2(3k+i\omega)^2 + (b_1+3b_3\hat{v}_e^{\,2})(k+i\omega) + \tfrac{3}{4}b_3\sigma^2(3k+i\omega)$$
$$+ (c_1+3a_3\hat{v}_e^{\,2} + \tfrac{3}{4}c_3\sigma^2) = 0 \qquad (31)$$

This equation may be compared with that obtained by local linearisation, using $\hat{v} = \hat{v}_e + \hat{v}'$ and neglecting $O(\hat{v}'^2)$, so that seeking a solution of the form $\hat{v}' = \alpha_v e^{\mu t}$ gives

$$\mu^3 + [a_1+3a_3\hat{v}_e^{\,2}]\mu^2 + [b_1+3b_3\hat{v}_e^{\,2}]\mu + [c_1+3c_3\hat{v}_e^{\,2}] = 0, \qquad (32)$$

where $\mu = \lambda + i\nu$. The linear stability boundaries are defined by $\lambda = 0$, yielding the critical equilibrium state $\hat{v}_b$, given by either

$$c_1+3c_3\hat{v}_b^{\,2} = 0 \text{ for real roots with zero damping} \qquad (33)$$

or $$[a_1+3a_3\hat{v}_b^{\,2}][b_1+3b_3\hat{v}_b^{\,2}] - [c_1+3c_3\hat{v}_b^{\,2}] = 0, \text{ for complex roots with zero damping, ie } \mu = i\nu \qquad (34)$$

The corresponding nonlinear conditions of $k(\sigma) = 0$ give the amplitude σ_C at which divergence occurs if the corresponding $\omega(\sigma_C)$ is imaginary, and the amplitude σ_L of any limit cycles, for real $\omega(\sigma_L)$. The equations are:

$$c_1 + 3c_3\hat{v}_e^{\,2} + \tfrac{3}{4}c_3\sigma_C^{\,2} = 0 \qquad (35)$$

and $$[a_1+3a_3\hat{v}_e^{\,2} + \tfrac{3}{4}a_3\sigma_L^2][b_1+3b_3\hat{v}_e^{\,2} + \tfrac{3}{4}b_3\sigma_L^{\,2}] - [c_1+3c_3\hat{v}_e^{\,2} + \tfrac{3}{4}c_3\sigma_L^{\,2}] = 0 \qquad (36)$$

It may be observed that these conditions are satisfied if

$$\hat{v}_b^{\,2} = \hat{v}_e^{\,2} + \tfrac{1}{4}\sigma_C^{\,2} \quad\text{or}\quad \hat{v}_b^{\,2} = \hat{v}_e^{\,2} + \tfrac{1}{4}\sigma_L^{\,2} \qquad (37)$$

Thus there appears to be a relationship between the amplitude of the limit cycle, or the critical amplitude, about an equilibrium state $\hat{v}_e$, and the amplitudes of the linear stability boundaries. In particular, the value of σ_L at $\hat{v}_e = 0$ (for responses about the zero equilibrium state) is twice that of the amplitude of the equilibrium state at which the linearised equations indicate zero damping. This relationship has been confirmed for this mathematical model, and the results are described in Section 3.1. For other nonlinear dynamic systems, the relationship, if it exists, will depend on the form of the nonlinearity, since equations (37) apply for cubic terms only.

The corresponding solutions for the variables p and r are found to be of the form $p = p_e + \alpha_p\sigma\cos(\phi+\epsilon_p)$, $r = r_e + \alpha_r\sigma\cos(\phi+\epsilon_r)$, where the amplitude ratios, $\alpha_p = |p/\hat{v}|$, $\alpha_r = |r/\hat{v}|$ and phase angles ϵ_p, ϵ_r are approximately independent of the amplitude of $\hat{v}'$. This approximation has been found to be justified for systems with nonlinearities in one variable, but may not be valid if other nonlinearities are included.

An attempt has been made by Simpson[18] to apply the averaging technique directly to the simultaneous differential equations of motion, such as equations (26), (27) and (28), without reducing the equations to one variable. This method, used for the current example, results in a stability determinant,

$$\begin{vmatrix} y_{v1} + 3y_{v3}\hat{v}_e^{\,2} + \tfrac{3}{4}y_{v3}\sigma^2 - \mu & \hat{w}_o & \hat{u}_o \\ l_{v1} + 3l_{v3}\hat{v}_e^{\,2} + \tfrac{3}{4}l_{v3}\sigma^2 & l_p - \mu & l_r \\ n_{v1} + 3n_{v3}\hat{v}_e^{\,2} + \tfrac{3}{4}n_{v3}\sigma^2 & n_p & n_r - \mu \end{vmatrix} = 0 \qquad (38)$$

The expansion of the determinant gives a stability polynomial for $\mu = k+i\omega$, viz

$$\mu^3 + [a_1+3a_3\hat{v}_e^{\,2}+\tfrac{3}{4}a_3\sigma^2]\mu^2 + [b_1+3b_3\hat{v}_e^{\,2}+\tfrac{3}{4}b_3\sigma^2]\mu + [c_1+3c_3\hat{v}_e^{\,2}+\tfrac{3}{4}c_3\sigma^2] = 0 \qquad (39)$$

The equations (39) and (31) are identical only for k = 0, ie for the study of stability boundaries, and work is in progress to study the rate of growth of limit cycles, given by $k(\sigma)$, to see which is the better approximation. The difference in the equations is due, in effect, to neglecting the terms arising from $\frac{d}{dt}(\hat{v}^3)$ in deriving equation (38). Such terms are implied when the stability determinant is expanded to give the stability polynomial, and it may be possible to retain them by extending the idea of a differential operator to certain classes of nonlinear equations. This type of procedure appears to be required for the approximate solution of equations of the type considered in Section 2.2, equation (18). The second-order terms do not contribute directly when the averaging process is applied, since $\frac{1}{2\pi}\int_0^{2\pi}\cos^3\phi\,d\phi = 0$ etc.

Thus the nonlinear stability determinant, for motion about the steady state $\{x_e\}$, of amplitude vector $\sigma\{\alpha\}$ is found to be $$|F(\underline{x}_e + \underline{\alpha}, \underline{c}_e) - \mu I| = 0 \qquad (40)$$
This is independent of the actual amplitude, defined by σ, (but dependent on the amplitude ratios $\{\alpha\}$), and so does not appear to predict nonlinear characteristics, where μ is a function of σ. It is hoped to study this problem in future work, and it may be noted that a similar approach[19] has been used for the determination of limit cycles in multivariate systems, ie when equations corresponding to (31) and (39) are identical in form for k = 0.

A different type of nonlinearity, viz hysteresis, is likely to be needed to express aerodynamic forces and moments when the airflow is separating and re-attaching due to the varying magnitudes of the response variables, particularly angle of attack. It is possible to evaluate the integrals involved in the averaging process, equations (30), for simple forms of hysteresis, but the technique has not been applied to problems in flight dynamics as yet.

3 EXAMPLES IN FLIGHT DYNAMICS

3.1 Lateral Responses with Cubic Nonlinearity in Moments due to Sideslip

The mathematical model used in Section 2.3 was originally developed to study the wing-rock phenomenon, but it has also demonstrated some more general properties, which were described briefly in Ref 6. Various nonlinear phenomena can occur, depending on the level of roll-damper gearing, $K_{\xi p}$ as illustrated in Fig 1. A limit cycle occurs for small values of $K_{\xi p}$ and for small disturbances, but larger disturbances lead to an exponential type of divergence. With increase of $K_{\xi p}$, the limit cycles do not occur, and the response after a small initial disturbance is a damped oscillation, but a larger initial disturbance leads to a divergent oscillation. The boundaries shown in Fig 1 have been derived using the approximate analysis of Section 2.3, for the zero equilibrium state, and confirmed by simulations on an analogue computer. A further study of the responses about non-zero equilibrium states has now been completed[20], and the relationship between the results from local linearisation of the equations of motion and from the approximate nonlinear analysis has been demonstrated.

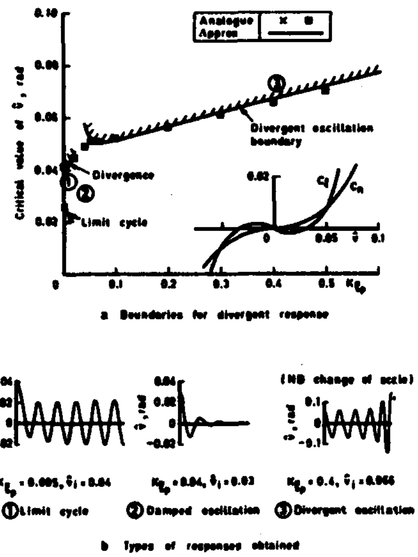


Fig 1. Characteristics of responses with cubic nonlinearities in sideslip and a roll-damper

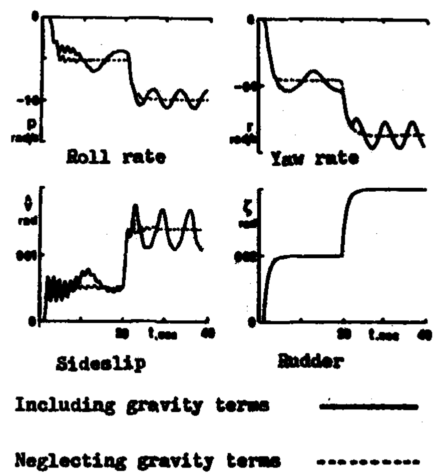


Fig 2. Effect of gravity terms on responses

The assumption that the gravity terms do not have a significant effect on the basic stability parameters has been shown to be justified by computing the responses using the complete equations of motion given in equations (26), (27) and (28). The gravity terms introduce an oscillation of constant amplitude, and frequency equal to the mean roll rate about the equilibrium states as shown in Fig 2.

The linear stability boundaries, obtained for gravity terms neglected, are shown as dashed lines on Fig 3, in terms of the bifurcation line, $\hat{v}_b$. For small roll damper gearing, $K_{\xi p}$, and small equilibrium state, the response is a divergent oscillation (unstable, Fig 3a), which becomes damped when the magnitude of the equilibrium state is increased beyond $\hat{v}_b$. At higher values of $K_{\xi p}$, the response about small equilibrium states is stable, but increase beyond $\hat{v}_b$ would lead to a divergent oscillation. The exponential mode also exhibits zero damping, (Fig 3b), and the linear boundary, $\hat{v}_b$, indicates the amplitude of the equilibrium state having a neutrally stable (constant) mode.

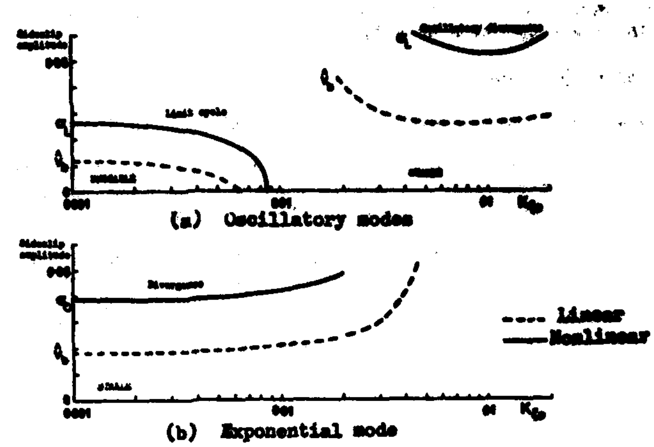


Fig 3. Comparison of linear and nonlinear stability boundaries

The nonlinear analysis gives the boundaries shown as solid lines in Fig 3 (replotted from Fig 1), and for this analysis about the zero equilibrium state, it is possible to include the gravity terms. It is readily seen that the "critical" amplitudes are approximately twice those obtained from the linear analysis, as implied by equation 37. Computed time histories confirm this relationship, for the divergences shown in Fig 4a, and more convincingly from results obtained at Cranfield[21] in Fig 4b for the amplitude of the limit cycle about different equilibrium states. Thus for this type of nonlinearity, (confined to one state variable), the local linearisation technique for responses about the non-zero equilibrium state, and the nonlinear analysis of responses at the zero steady state give related information in terms of critical amplitudes of the responses.

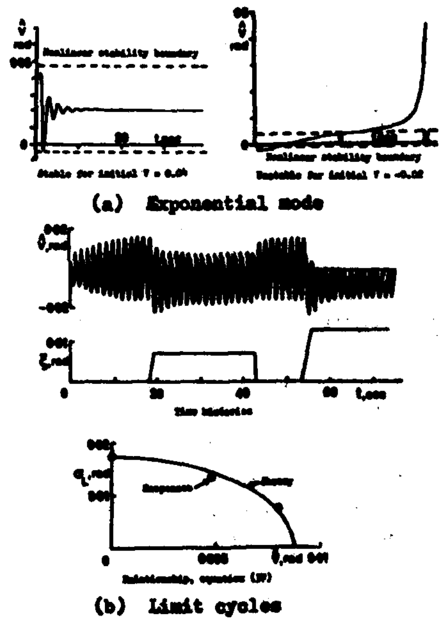


Fig 4. Responses about non-zero equilibrium states

3.2 Inertia Cross-Coupling

For nonlinear dynamic systems, it is possible for non-zero equilibrium states to exist which are independent of control settings. If the mathematical model is written in the matrix form of equations (13) and (18), then the condition for these autorotational states is

$$\left| A(\underline{x}_a) \right| = 0.$$

The solution of the determinantal equation for linear and some second-order aerodynamic forces and moments has been discussed in Section 2.1 and gives the auto-rotational states associated with inertia cross-coupling. Linear stability analysis about these states determines the initial type of response, in particular whether a stable response is possible. An extensive study has been made by Mehra[5] for three aircraft, one of which has second-order derivatives of the form given in equation (12). In the notation of Section 2.2, the matrices defined in equations (18) to (21) for this mathematical model are:

$$A_o = \begin{bmatrix} y_v & 0 & \sin\alpha_o & 0 & -\cos\alpha_o \\ 0 & z_w & 0 & \cos\alpha_o & 0 \\ l_v & 0 & l_p & l_q & l_r \\ 0 & m_w & 0 & m_q & 0 \\ n_v & 0 & n_p & 0 & n_r \end{bmatrix}$$

$$A_1(\underline{x}_e) = \begin{bmatrix} 0 & p_e & 0 & 0 & 0 \\ -p_e & 0 & 0 & 0 & 0 \\ l_{vw}\hat{w}_e & 0 & 0 & 0 & l_{rw}\hat{w}_e + b_x q_e \\ 0 & 0 & 0 & 0 & b_y p_e \\ 0 & n_{pw} p_e & 0 & b_z p_e & 0 \end{bmatrix}$$

$$B_o = \begin{bmatrix} 0 & y_\xi & 0 \\ z_\eta & 0 & 0 \\ 0 & l_\xi & l_\zeta \\ m_\eta & 0 & 0 \\ 0 & n_\xi & n_\zeta \end{bmatrix}$$

$$A_2(\underline{x}_e) = \begin{bmatrix} 0 & 0 & \hat{v}_e & 0 & 0 \\ 0 & 0 & -\hat{w}_e & 0 & 0 \\ 0 & l_{vw}\hat{v}_e + l_{rw} r_e & 0 & b_x r_e & 0 \\ 0 & 0 & b_x q_e & 0 & 0 \\ 0 & 0 & b_z r_e + n_{pw}\hat{w}_e & 0 & 0 \end{bmatrix}$$

$$B_2(\underline{c}_e) = \begin{bmatrix} 0 & 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 & 0 \\ 0 & 0 & l_{\xi w}\xi_e & 0 & 0 \\ 0 & 0 & 0 & 0 & 0 \\ 0 & 0 & n_{\xi w}\xi_e & 0 & 0 \end{bmatrix} \tag{42}$$

The autorotational states satisfy the determinantal equation

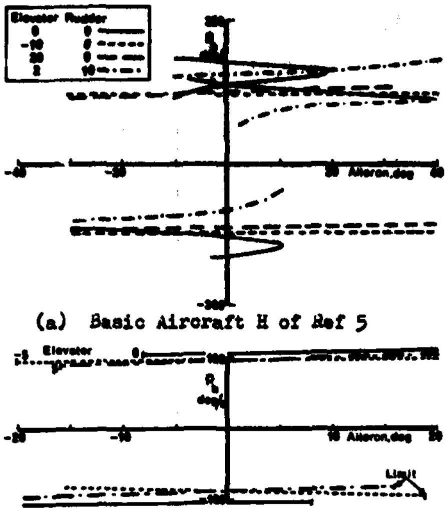
$$\left| A_o + A_1(\underline{x}_a) \right| = 0, \tag{43}$$

and their stability is given by the roots of the equation

$$\left| A_o + A_1(\underline{x}_a) + A_2(\underline{x}_a) - \mu I \right| = 0 \tag{44}$$

Thus, autorotational states of low magnitude can be expected to have near-zero roots, since the matrix $A_2(\underline{x}_a)$ does not contain roll rate (which is the dominant parameter), and so does not affect the eigen values of the matrix significantly.

The equilibrium states, for non-zero control settings, have stability characteristics defined by equations (22) and (23), and the bifurcation surfaces with zero damping can be expected to be near any autorotational states of small magnitudes. This is demonstrated in Mehra's results, as shown in Fig 5a. The autorotational states are found to be unstable at $p_a \approx \pm 180^o$/sec, and stable at $p_a \approx \pm 400^o$/sec, whereas the bifurcation surfaces for various combinations of control settings all lie near $p_b = \pm 180^o$/sec. The control laws designed by Mehra cannot change the magnitudes of the autorotational states, and the resulting bifurcation surfaces are again very close to $p_b = \pm 180^o$/sec (Fig 5b). These characteristics are thus largely independent of control setting, and so it seems advisable in future work to investigate first the stability characteristics as functions of the equilibrium state variables, and then check whether critical equilibrium states are achievable with the control powers available.



(a) Basic Aircraft H of Ref 5

(b) With aileron-rudder interconnect

Fig 5. Zero damping of linear modes (Bifurcation surfaces)

3.3 Spinning Motion

The ideas suggested by Pinsker[10] for a unified approach to the determination of autorotational rolling and spinning states can be applied to the forms of mathematical model discussed in Section 2.1, and work is continuing to investigate the relative accuracies of different approximations. Preliminary work indicates that it may be possible to express stability criteria in terms of the equilibrium states for some of the usual forms of aerodynamic nonlinearity occurring at very high angles of attack, so it is hoped to explore this concept further.

4 EXPERIMENTAL AND THEORETICAL RESEARCH PROGRAMME

A research programme is in progress at the RAE[8], using wind-tunnel and free-flight models of a research configuration to provide experimental data to be used in the investigation of mathematical modelling at high angles of attack. Although the form of the mathematical model is not being restricted to types which are amenable to the analysis described in Section 2, the opportunity is being taken to test various approximations for the prediction of flight characteristics.

The research configuration is shown in Fig 6. The basic research programme entails tests at low speed, but the design of the model has also been influenced by the need for satisfactory characteristics

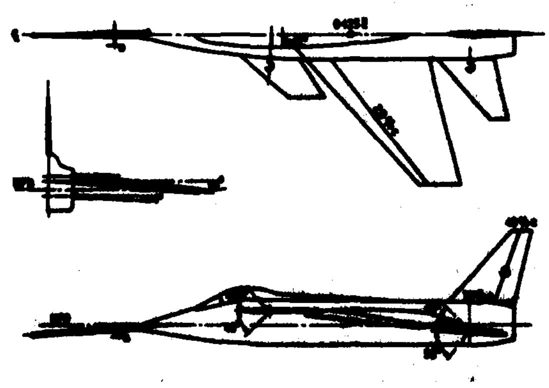


Fig 6. Geometry of the High Incidence Research Model.

at transonic speeds, so the wing has a modern supercritical section, and planform to give good performance. A canard configuration was chosen, being typical of new designs for high performance, but a separate tailplane was retained as the aft control for ease of manufacture. A single fin with rudder was sized to give adequate directional stability at moderate angles of attack. The canard and tailplane surfaces may both be moved differentially for roll control and sideforce control if required.

Analysis of the static wind tunnel tests is nearing completion[22], and the results indicate that the free-flight models should be controllable up to $\alpha=30^\circ$ with a simple on-board stability augmentation system. Wind-tunnel tests using the oscillatory rig at RAE have been completed, and the model is also being used on the rotary rig at BAe Warton. Current work is to establish mathematical models at various levels of approximation, such as linearised aerodynamic derivatives, then nonlinear static characteristics due to angles of attack and sideslip, then nonlinear rotary characteristics due to roll rate, and also to establish a numerical data base for representation of the basic results from the wind tunnels. All of these mathematical models will be used to predict flight behaviour, obtaining both computed responses and calculated stability characteristics. This information will then be used to design various active control systems, and to design the free-flight experiments.

The free-flight models are being built at the 2.25 scale of the definitive wind-tunnel model, and it is planned to conduct preliminary trials in the UK later in 1982, preparatory to the main series of trials to be conducted in the US jointly with NASA Dryden in 1983. Several types of control inputs will be applied, designed to give responses suitable for deriving aerodynamic data using parameter identification techniques, or for establishing departure boundaries, or for demonstrating spin characteristics and spin recovery, or for testing the active control system. The models will be flown initially with about 5% static stability, but a relaxed stability system will also be tested with about 5% static instability, up to and beyond departure from controlled flight. There should also be sufficient data for some analysis using system identification techniques, to establish the form of mathematical model needed to represent departure characteristics adequately. The predictions using nonlinear mathematical models should give some insight to the types of nonlinearity which need to be represented in order to give the observed departure characteristics, and so provide some guidelines for the system identification.

5 CONCLUSIONS

The application of two analytical techniques developed to determine the stability characteristics of nonlinear equations of motion of aircraft has demonstrated that results can be expressed in terms of the magnitudes of the response variables for some particular problems. The stability boundaries are thus not directly dependent on control settings, and so may be interpreted generally for all types of manoeuvres. The nonlinear stability boundaries also indicate which type of departure may be encountered. A research programme is briefly described, which will provide data from wind-tunnel and free-flight experiments to be used in the prediction and analysis of nonlinear phenomena at high angles of attack.

LIST OF REFERENCES

1. W.P. Gilbert, L.T. Nguyen and R.W. van Gunst. Simulator study of the effectiveness of an automatic control system designed to improve the high angle of attack characteristics of a fighter airplane. NASA TN D-8176. 1976.

2. Anon. Dynamic Stability Parameters, AGARD CP 235, 1978 and AGARD LS 114, 1981.

3. J.R. Chambers and S.B. Grafton. Aerodynamic characteristics of airplanes at high angles of attack. NASA Tech. Memo. 74097, 1977.

4. R.F. Stengel and P.W. Berry. Stability and control of manoeuvring high-performance aircraft. NASA CR 2788, 1977.

5. R.K Mehra, W.C. Kessel and J.V. Carroll. Global stability and control analysis at high angles of attack. ONR CR215-248-1, 1977.

6. A. Jean Ross. Lateral stability at high angles of attack, particularly wing rock. Paper 10 of AGARD CP 260, 1978.

7. L. J. Beecham and I.M. Titchener. Some notes on an approximate solution for the free oscillation characteristics of nonlinear systems typified by $\ddot{x} + F(x, \dot{x}) = 0$. ARC R&M 3651, 1969.

8. G.F. Moss and A. Jean Ross. A programme of work on the flight dynamics of departure using a high incidence research model (HIRM). RAE Tech. Memo. to be published.

9. B.W. McCormick. Equilibrium spinning of a typical single-engine low-wing light aircraft. J. Aircraft, Vol 18, No. 3, 1981.

10. W.J.G. Pinsker. The theory of autorotational rolling and spinning. RAE TR 81076, 1981.

11. W.M. Adams Jr. Analytic prediction of airplane equilibrium spin characteristics. NASA TN D-6926, 1972.

12. P.J. Yangos and J.P. Yangos. Spin: angles and inertial moments. R.Aero. Soc. Aeronautical Journal, July/August 1981.

13. W.J. Duncan. The principles of the control and stability of aircraft. Cambridge University Press, 1959.

14. D.E. Johnston and J.R. Hogge. Non-symmetric flight influence on high angle of attack handling and departure. J. Aircraft, Vol. 13, No. 2, 1976.

15. M.A. Hreha and F.H. Lutze. Linear analysis of post-stall gyrations. J. Aircraft, Vol. 17, No. 10, 1980.

16. H.H.B.M. Thomas and P. Price. A contribution to the theory of aircraft response in rolling manoeuvres including inertia cross-coupling effects. ARC R&M 3349, 1964.

17. A. Jean Ross and L. J. Beecham. An approximate analysis of the nonlinear lateral motion of a slender aircraft (HP 115) at low speeds. ARC R&M 3674, 1971.

18. A. Simpson. An algorithm for autonomous nonlinear dynamical equations. Roy. Aero. Soc. Aeronautical Quarterly, August 1977.

19. J.H. Taylor. A general limit-cycle analysis method for multi-variable systems. Paper in New Approaches to nonlinear problems in dynamics. SIAM 1980.

20. A.Jean Ross and M. Johnson. A comparison between two analytical methods of studying the characteristics of nonlinear equations of motion of an aircraft. RAE TR to be published.

21. R. Thorne. PhD thesis to be published. Cranfield Institute of Technology.

22. G.E.A. Reid and A. Jean Ross. The development of mathematical models for the RAE High Incidence Research Model, Part I. Analysis of Static Aerodynamic Data, RAE TR to be published.

EFFECT OF CONTROL SYSTEM DELAYS ON FIGHTER FLYING QUALITIES*

by

Rogers E. Smith and Randall E. Bailey
Arvin/Calspan ATC
Flight Research Department
Buffalo, New York 14225

SUMMARY

The flying qualities problems of the latest fighter aircraft are most often related to the time delay which is introduced into the flight control system by the advanced, typically complex, control system design. The intent of this paper is to confirm that time delay has a significant effect on fighter flying qualities, both longitudinal and lateral. Existing flying qualities research data from in-flight simulators are used to demonstrate this point. Typical sources of flight control system time delay and the methods of time delay "measurement" are reviewed. Finally, the application of several candidate flying qualities evaluation criteria or requirements, which are applicable to highly augmented fighter aircraft, is discussed.

INTRODUCTION AND PURPOSE

Fighter aircraft with advanced flight control systems which essentially depend on electrical signals to communicate the pilot's commands to the control surfaces are a reality. Examples are the F-16, YF-17, F-18, Tornado and AFTI/F-16 aircraft. For today's advanced fighter aircraft, the capability exists to tailor flying qualities for diverse mission tasks through use of high-authority electronic augmentation systems. Unfortunately, the potential of this expanded technical capability has not been realized - in fact, new flying qualities problems have often been created in the process of solving the old ones.

These new flying qualities problems are most often related to the aircraft's initial delay in response to a pilot input. This delay is introduced by the advanced, and typically overly complex, flight control system design. The source of these time delays can be from the higher order complexity of the flight control system design or, in the case of digital systems, the inherent digital time delays. Digital flight control systems tend to be the worst offenders since the power of the computer unfortunately encourages the design of very complex control systems. For fighter aircraft, even apparently small time delays can cause a dramatic degradation in flying qualities for precision tasks.

The purposes of this paper are to:

- Review the sources of time delay in typical fighter flight control system designs and discuss the various methods which are used to "measure" the initial delay to a pilot input.
- Confirm the fact that time delay, either pure or equivalent time delay, is a significant flying qualities parameter.

  Evidence for this point is drawn from pertinent fighter flying qualities research data which includes longitudinal and lateral flying qualities data for fighter precision tracking (Flight Phase Category A) and landing (Category C) tasks.
- Discuss the available design criteria or specifications with which the effects of control system time delay on fighter flying qualities can be evaluated. In particular, the importance of relying only on data for realistic "highly stressed" precision tasks is emphasized.

WHAT IS A TIME DELAY?

The pilot of any aircraft, but particularly the pilot of a fighter performing precision tasks, wants a response to his stick input immediately. Any delay in the response to his input detracts from his ability to perform the task; it interferes with his ability to coordinate instinctively his brain (desired response), hand, and the observed aircraft response. His tolerance to delay in the response to his input has limits, particularly for precision tasks such as tracking, refueling, formation, and landing.

Modern fighter aircraft with advanced electronic flight control systems are especially vulnerable to problems related to excessive time delay which can cause serious flying qualities problems. This time delay can come from a variety of sources in the typical advanced flight control system.

---

*The NT-33 aircraft research work used in this paper was supported under contract by the United States Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, Ohio.

Sources of Time Delay

By definition, a system which reproduces the exact form of an input after a specific interval of time is defined as exhibiting transport time delay or pure time delay. In Laplace notation, pure time delay is expressed as $e^{-\tau s}$. The initial response delay of today's fighter aircraft to pilot commands is not, however, due solely to pure time delays. Additional delay in the aircraft's response occurs from sources which do not, by definition, exhibit pure time delays. The pilot is, of course, concerned with the overall delay of his input which is the sum of any pure time delay and the "equivalent" time delay from other sources. There are three sources of time delay which are the primary contributors to the total delay exhibited in the initial response of today's aircraft to pilot inputs.

- Digital Computational Delay:

  It is a source of pure time delay in flight control applications and represents a potential penalty to attain the advantages of the digital system; in general, this contribution to the overall delay is not significant.

- Sampling Delay:

  It is a part of a digital flight control system where the system input and response data must be sampled. The amount of pure time delay associated with this source is a function of the sampling rate and the time that the input is made. Typically the delay is assumed to be one-half the sampling interval (Reference 1). The differences between sampling rate delays and computational (pure) time delay are graphically illustrated in Figure 1.

- High Order System Equivalent Delay:

  It is the consequence of cascading dynamic elements in the flight control system and is the major cause of control system time delay. These elements may be related to the required filtering or to the design strategy of the flight control system.

  As a simple example, the response of a first-order lag filter to a step input is shown in Figure 2. Note that the response does not exhibit any time delay; however, cascading two first-order lag filters creates a finite time delay. Thus the addition of a first-order prefilter (and associated phase lag) can add time delay to a control system even though the element itself does not exhibit pure time delay.

This last source of delay is often referred to as "equivalent" time delay. While this modifier is descriptive, it is necessary to specify how the equivalent time delay is "measured". A review of the techniques typically used to measure time delay is therefore in order.

"Measures" of Time Delay

- Visual Time Delay

Visual, or "eyeball" time delay is the period of time for which the response to an input is essentially zero. For example, the delay in the response for the cascaded system in Figure 2 is obvious. In reality, the period of time during which responses are below the pilot's threshold of perception can be viewed as time delay. However, the measure of this visual time delay is not very precise since it is a function of the size of the input and the scale of the time history. It is, therefore, not a very useful method of characterizing the overall time delay of a control system.

- Equivalent Time Delay

The equivalent time delay of the flight control system can be "measured" by matching the frequency response of the complex system over a specific frequency range with a classic low order model. A simplified example is illustrated in Figure 3. For example, the constant speed pitch rate response low order model is:

$$\frac{q}{F_{ES}} = e^{-\tau_E s} \cdot \frac{M_{F_{ES}}(s+1/\tau_{\theta_{2_E}})}{s^2+2\zeta_E\omega_E s+\omega_E^2}$$

Where F_{ES} is the pilot pitch force input,

τ_E is the control system equivalent time delay,

ζ_E, ω_E, $1/\tau_{\theta_{2_E}}$ are the equivalent short period and $1/\tau_{\theta_2}$ parameters, and

$M_{F_{ES}}$ is the pitch control sensitivity.

Thus, the more complex pitch rate response of an advanced aircraft is reduced to a "classic" low order form using a parameter optimization process across a frequency range of interest. In this fashion, the equivalent control system time delay is measured (see References 2, 3 and 4 for details). A cost function is calculated to gauge the "goodness of fit" or degree of mismatch between the actual and low order model. To extract this measure a frequency response of the system is required; these data can be obtained rather precisely by judiciously perturbing the aircraft at discrete frequencies using automatic sine wave inputs (Reference 5). Alternatively, the application of Fast Fourier Transformations to flight data generated by a variable frequency sine wave-type inputs has been proposed for military specification compliance (Reference 6). This equivalent time delay is a measure of the delay from pilot input to aircraft response. Since the model assumed has no initial response delay, the equivalent delay is also a measure of the delay from pilot input to the control surface. Equivalent systems matching of control system elements alone has also been performed (Reference 7).

Flight control system elements with natural frequencies significantly higher than the dominant airframe natural frequencies introduce only phase distortion and produce time delay without affecting the equivalent frequency of the response. The effect of low frequency elements is to distort both the phase and amplitude in the frequency range of interest and both equivalent delay and frequency are affected.

- Effective Time Delay:

Effective (to distinguish from equivalent) time delay is derived directly from a time history response by the maximum slope intercept method. This measure is the difference in time between the application of a step input and the intersection of the maximum slope tangent to the response (Figure 4). The effective time delay measure (Reference 8 to 10) does not require an assumed low order model but the value of delay calculated is a direct function of how the initial response differs from a pure first-order lag-type time response.

Comparison of Time Delay "Measures"

For flight control systems with high frequency dynamic elements, the time delay measured using either the maximum slope method (time domain) or the equivalent system method (frequency domain) are essentially the same. The two methods, however, produce different time delay measures when low frequency dynamic elements are present in the flight control system.

Since today's fighter aircraft have flight control system designs which can have both high and low frequency dynamic elements, the measures made by the two techniques are not typically interchangeable. Flying qualities design guidelines or specifications based on time delay measures must therefore be applied with this caution in mind. The time delay measurement technique must be clearly specified. To illustrate these points consider Configuration 5-3 and 5-3F1 from the NT-33 Lateral Higher Order System (LATHOS) program (Reference 10).

The simplified roll rate to roll stick force transfer functions of interest are:

- 5 - 3 $\quad \dfrac{p}{F_{AS}} = \dfrac{(p/F_{AS})_{ss}}{(0.15\,s + 1)(0.025\,s + 1)}$

- 5 - 3F1 $\quad \dfrac{p}{F_{AS}} = \dfrac{(p/F_{AS})_{ss}}{(0.15\,s + 1)(0.1\,s + 1)}$

where $(p/F_{AS})_{ss}$ is the steady state roll rate per pound

Each configuration transfer function also included a 60 rad/sec second order actuator and a 200 rad/sec first order system filter which are not shown for clarity but which are included in the time histories and time delay calculations in Figures 5 and 6. The only difference between the two configurations was the increase in the roll prefilter time constant from .025 to .10 sec. The new prefilter smoothed the initially abrupt roll response of 5-3 (Pilot Rating, PR = 7) and produced a satisfactory aircraft (PR = 3 for 5-3F1).

For Configuration 5-3 in which the control system dynamic elements are all essentially high frequency, the calculated time delay and roll mode time constant are nearly identical by both methods (Figure 5).

For Configuration 5-3F1 in which the prefilter affects both the phase and amplitude in the frequency range near that of the airframe roll mode time constant, the time delay is appreciably different from the two methods (Figure 6). Note that the equivalent or effective roll mode time constant has changed due to the prefilter. In this case, the flying qualities cannot be related to the time delay increase alone since the transient response of the aircraft has also been changed. More details on the extraction and interpretation of these parameters are presented in the major section on Evaluation Criteria.

EFFECTS OF TIME DELAY ON LONGITUDINAL FLYING QUALITIES

The purpose of this section is to confirm that control system time delay can have a significant effect on fighter longitudinal flying qualities for both Flight Phase Category A (tracking) and C (landing) tasks. Substantiating data are drawn from recent flying qualities research programs using the USAF NT-33 research aircraft operated by Calspan and the NASA Digital Fly-by-Wire (DFBW) F-8 aircraft. These data are specific to fighter aircraft performing realistic precision fighter tasks. The criticality of the task in the evaluation of aircraft with control system delays is discussed further in a separate major section.

Approach and Landing Tasks (Category C)

The effect on flying qualities of adding delay to the pitch control system of an otherwise good aircraft is illustrated in Figure 7. Sources of the data on the figure are:

- NT-33 Landing Approach Higher Order System (LAHOS) Program (Reference 11) and McDonnell-Douglas "McFit" equivalent system parameters for the data (Reference 12),
- NT-33 Equivalent System Program (ESP) (Reference 13),
- NT-33 PIO Suppressor Program (PIOS) (Reference 14),
- F-8 Approach and Landing Program (Reference 15).

For the first NT-33 program (Reference 11), the additional delay was in the form of higher order dynamic elements for which the equivalent delay was "measured" using the "McFit" frequency domain method (Reference 2). For the cases selected, essentially the same time delays would be extracted by the maximum slope method. The other NT-33 programs (References 13 and 14) utilized a combination of pure and equivalent time delay; the total of the two delay sources is used in Figure 7. Finally, the data from the F-8 is based on the addition of pure digital time delay beyond a threshold basic value of 130 milliseconds (ms). In each case the equivalent delay includes all the delay of the control system elements. The constant speed pitch rate transfer function is of the form:

$$\frac{q}{F_{ES}} = e^{-\tau_E s} \cdot \frac{M_{F_{ES}}\,(s+1/\tau_{\theta_2})}{s^2+2\zeta_E\omega_{sp}s+\omega_E^2}$$

where, F_{ES} is the pilot pitch force input,

τ_E is the total equivalent control system time delay; the values of ζ_E and ω_E are essentially constant for Level 1 for each set of data and $1/\tau_{\theta_2}$ is the actual aircraft value, and

$M_{F_{ES}}$ is the pitch control sensitivity.

Summary observations from the longitudinal approach and landing task data are:

- Equivalent time delay greater than a threshold of approximately 130 ms significantly degrades longitudinal flying qualities.
- The flying qualities pilot rating degradation, considering all the data used, is approximately 1 PR for 25 ms of equivalent time delay.
- Degradation is similar whether the source of delay is pure time delay or equivalent delay from higher order control system elements or some combination of the two delay sources.
- The results from the LAHOS program (Reference 11) for the very low short period damping ratio cases is also of particular interest. For the low damping ratio cases without initial delay, the pilot was able to land without any Pilot Induced Oscillation (PIO) problems. Even though the overall response was poor, he was able to develop a consistent control strategy because the response to his input was immediate. With time delay present in an aircraft, the tie between input and response is broken which can lead to overcontrol and PIO's in precision tasks.

Tracking and Formation Tasks (Category A)

Unfortunately, a data base similar to that used for the pitch approach and landing time delay discussion does not exist. Either the incremental equivalent time delay was added in a fashion which also changed the equivalent short period frequency significantly or precision tracking tasks were not performed.

Data sources from which some insight can be gained are:

- NT-33 Neal-Smith Program (Reference 16),
- F-8 Formation Data (Reference 15).

For the Neal-Smith program, the additional delay was created by cascading dynamic elements. Equivalent system parameters, including an equivalent time delay, τ_E, can be extracted from the data base using the "McFit" approach. Unfortunately, the additional control system dynamics typically added an equivalent time delay and changed the equivalent short period frequency. Therefore, for this data source, the effect of time delay alone cannot be easily isolated.

Hodgkinson in Reference 2 used regression analysis to correlate the Neal-Smith pilot rating data and calculate a sensitivity to equivalent time delay of approximately 1 PR degradation per 50 ms of delay. Using the data presented in Reference 6 where the Neal-Smith data were correlated using equivalent time delay and bandwidth frequency, estimates of pilot rating sensitivity to time delay of 1 PR per 25-30 ms can be made using some imagination.

The results from the time delay studies using the NASA DFBW F-8 aircraft performing precision formation tasks indicated that the sensitivity to time delay beyond a threshold of 130 ms was 1 PR degradation per 40 ms of delay. Investigations of time delay effects on fighter lateral flying qualities (Reference 10) showed that the target tracking task was significantly more demanding than the formation task. On this basis, it might be expected that a study using a real pitch tracking task would produce a steeper flying qualities degradation with time delay which is closer to the approach and landing pitch task results.

Summary observations from these limited longitudinal tracking and formation task data are:

- Equivalent time delay greater than a threshold of approximately 130 ms significantly degrades pitch flying qualities.
- The flying qualities degradation, which can only be estimated from the data available, is approximately 1 PR per 30 ms of equivalent time delay. It is therefore estimated to be approximately the same for both A and C Flight Phase Category tasks. More data are, however, required for fighter tracking tasks.

EFFECTS OF TIME DELAY ON LATERAL FLYING QUALITIES

The purpose of this section is to confirm that control system time delay can have a significant effect on fighter lateral flying qualities for both Flight Phase Category A (tracking) and C (landing) tasks. Data for this confirmation is drawn from a recent flying qualities research program using the USAF NT-33 aircraft which was specific to fighter aircraft performing realistic precision fighter tasks.

Approach and Landing Tasks (Category C)

The effect on flying qualities of adding delay to the lateral control system of an otherwise good aircraft is illustrated in Figure 8. Source of the data on the figure is:

- NT-33 Lateral Higher Order System (LATHOS) Program (Reference 10).

In this program, a combination of pure and equivalent time delay was systematically added to the lateral flight control system; the total delay is used in Figure 8. The equivalent delay portion would be "measured" as the same value using either the "McFit" or maximum slope methods. Assuming the Dutch roll mode is effectively cancelled and the spiral is neutral, the roll rate to roll stick force transfer function is of the form:

$$\frac{p}{F_{AS}} = e^{-\tau_E s} \cdot \frac{L'_{F_{AS}}}{(s+1/\tau_{R_E})}$$

Where, F_{AS} is the pilot force input,

τ_E is the total equivalent control system time delay, and the value of τ_{R_E} the equivalent roll mode time constant, are essentially constant and Level 1 for the data set, and

$L'_{F_{AS}}$ is the roll control sensitivity.

Summary observations from this limited lateral approach and landing task data are:

- Equivalent time delay greater than a threshold of approximately 120 ms significantly degrades lateral flying qualities.
- The flying qualities degradation, considering all the data is estimated to be 1 PR per 30 ms of equivalent delay.
- The trends exhibited are similar to the longitudinal results for the approach and landing task.
- The data base for this flight phase is small; more data are required.

Other data in Reference 13 show a much greater tolerance for lateral time delay than indicated by the data in Figure 8. A time delay threshold of approximately 220 ms is reflected by the Reference 13 data; the degradation rate is similar to that of Figure 8 beyond the threshold. The data in Figure 8 were obtained using very realistic aggressive fighter offset landing tasks. It is hypothesized that the somewhat less aggressive task used in the admittedly preliminary research program of Reference 13 may be the source of this large difference in delay threshold value.

Tracking, Refueling and Formation Tasks (Category A)

The effect on the flying qualities of an otherwise good aircraft of adding delay to the roll control system is also illustrated in Figure 8. Source of the data on the figure is:

- NT-33 Lateral Higher Order System (LATHOS) Program (Reference 10).

The same ground rules for the data described in the last subsection apply to this discussion. Tasks for this flight phase in the experiment included actual tracking, refueling and formation tasks. The data base gathered during this program was more extensive than shown in Figure 8 and also included a study of the effects of combinations of low-frequency prefilter and equivalent time delay. These data are discussed in the major section addressing flying qualities evaluation criteria; the results used in Figure 8 are restricted to the effects of time delay alone on an otherwise good aircraft.

Summary observations from these limited tracking task data are:

- The same threshold and degradation trend shown for the lateral approach and landing data can be estimated.

This brief review of pertinent fighter flying qualities data has confirmed that control system delay, in the form of pure transport delay or an equivalent delay from cascaded high frequency elements, is a very significant flying qualities parameter. Obviously, the overall flying qualities of a fighter aircraft are related to many parameters not just time delay.

Before the candidate methods for evaluating the flying qualities of fighter aircraft with advanced, complex, delay-prone flight control systems are discussed, it is important to review the effects of task on the evaluation of these aircraft.

TIME DELAY, PILOT TECHNIQUE, AND THE TASK

The evaluation of highly augmented fighter aircraft with appreciable time delay in the initial control surface response to a pilot input is very much a function of pilot technique and the degree of precision demanded by the task. For example, the flying qualities of an aircraft with appreciable initial delay may be satisfactory for the approach phase of the landing task but deteriorate dramatically near touchdown as the required task precision increases.

The results from the NASA F-8 research program (Reference 15) clearly illustrate the significant effect of task performance standard on the flying qualities effects of pitch time delay (Figure 9).

- For the high stress pitch landing task, which included a lateral offset maneuver and a specific touchdown zone, the degradation in PR is much steeper than for the low stress task. Also shown on Figure 9 are the NT-33 data (Reference 11) for a similar task which correlates well with the F-8 high stress data.

- The low stress task involved a straight-in approach with no touchdown zone constraints; the data trends are similar to those obtained in a sophisticated fixed base simulator using the task and configurations from the NT-33 program reported in Reference 11.

- An important point to be made is that realistic stress levels cannot be properly replicated in ground based simulators. Flying qualities evaluations of aircraft with complex, delayed responses to pilot inputs cannot, therefore, be reliably conducted on ground simulators.

The effect of pilot technique on the evaluation of fighter aircraft with significant control system time delay can best be illustrated using an excerpt from the discussion of the results in Reference 13, the NT-33 approach and landing program to study equivalent systems.

"Previous flying qualities studies (References 11 and 17, for example) have indicated that, for aircraft with significant control system dynamics, small variations in pilot technique or task performance standard can result in dramatic variations in the pilot rating data. These aircraft have been appropriately described as having lurking 'flying qualities cliffs'. The results from this experiment also have examples of significant variations in rating between evaluation pilots. Before any attempt is made to analyze the data, the following information should be considered.

Pilot A, who was the primary evaluation pilot for the overall program, worked very hard to maintain a constant standard of task performance despite, in some cases, the obviously poor flying qualities of a particular configuration. His flying technique was observed to be representative of typical fighter pilots. In contrast, the other main evaluation pilot, Pilot B, sometimes demonstrated very specialized pilot techniques when flying PIO prone aircraft. He is an exceptionally smooth and predictive pilot. However, when 'backed into a task corner', i.e., when he got into a situation where he couldn't use his adaptive technique, his performance was similar to that of Pilot A who tended to fly in a more continuous closed-loop fashion.

Consider Pilot B's evaluation of Configuration P12 (Flt. 2073): This evaluation is a classic example of the problems involved in flying qualities evaluations of marginal highly augmented aircraft. Special techniques or task conditions can allow the aircraft to 'pass' the evaluation but, when exposed to the real world environment and placed in the hands of an 'average' pilot, the 'failure' can be disastrous. During the evaluation in question, Pilot B flew the first two landings with no real difficulty apparent - he was able to preplan his task and fly smoothly and predictively. On the third approach he inadvertently allowed the sink rate to get too high, too close to the ground; urgent action was required to prevent a very hard landing. The result: a full stall, 10 feet above the runway - he overcontrolled badly because of the large time delay in the pitch control system. When forced into a tight task his performance was the same as Pilot A who had rated the configuration a 9.

Unfortunately, he blamed himself not the evaluation aircraft and, after flying another approach and landing in which he was able to return to his predictive landing technique, he gave the aircraft a 5 rating."

These flying qualities research examples illustrate that the evaluation of today's fighter aircraft with complex, delay-prone, flight control systems is not easy. The task and the standard of performance used in the task, as well as the technique of the pilot, are critical factors; careful control of these factors are necessary for a valid evaluation.

Examples from specific aircraft development programs also demonstrate these points.

- F-18A: Initial design proved to be PIO-prone at touchdown due to excessive initial time delay in pitch (Reference 17); these flying qualities problems were only exposed when evaluated in the high stress, realistic environment provided by an in-flight simulator.
- Tornado: Serious pitch landing flying qualities problems were exposed mid-way through the real aircraft flight test program (Reference 18); pilot technique and task conditions exposed a "cliff" which was related to excessive initial pitch time delay.
- YF-17: A serious pitch-landing PIO problem was exposed on the NT-33 in-flight simulator which was not evident during ground based simulator evaluations (Reference 19). The cause of the problem was excessive initial delay introduced by a low frequency prefilter.
- Space Shuttle: Although clearly not a fighter aircraft, the results of the initial free-flight tests (Reference 20) are pertinent to this discussion. Serious pitch PIO problems, caused in part by excessive initial response delay, emerged when the task became more stringent. The problems occurred on the last preflight test where the task was to land on a real runway rather than the huge lake bed landing site used in the first four tests.

In summary, we have confirmed that time delay is a critical factor in fighter flying qualities and established that valid evaluations require careful attention to the task details and pilot technique. Potential flying qualities evaluation criteria which are applicable to fighter aircraft with complex, delay-prone, flight control systems are reviewed in the next section.

FLYING QUALITIES EVALUATION CRITERIA

For typical advanced fighter digital flight control designs, the initial response delay is made up of a combination of pure transport time delay and equivalent or effective (depending on the "measurement" method) time delay. The additional control system dynamics, or high order dynamics, consist of both high and low frequency elements. High frequency elements (high relative to the frequency of the principal short-term response mode) introduce time delay but do not change the response shape; low frequency elements contribute time delay (phase distortion) and change the shape of the response (amplitude distortion).

A flying qualities requirement or design criteria must, therefore, deal with the effect of the flight control system on the overall aircraft response to a pilot input. The dimensions of the criterion must include both the time delay (pure and equivalent) and the important equivalent aircraft response parameters.

The purpose of this section is to comment on several existing flying qualities evaluation criteria or requirements applicable to aircraft with time delay and, in particular, to present the time history criterion from the NT-33 Lateral Higher Order System Program (Reference 10).

U.S. Military Flying Qualities Specification, MIL-F-8785C (Reference 21)

The present military flying qualities specification, MIL-F-8785C, addresses the evaluation of highly augmented fighter aircraft through the equivalent system method. Unfortunately, the type of equivalent system matching process is not specified, which diminishes the usefulness of the specification. Time delay is addressed through specific allowable delay values and indirectly through a requirement which limits the control system surface phase lag at particular equivalent frequencies. Again, the lack of suitable definitions for the equivalent delay measurement leaves the requirement somewhat ineffective.

The preliminary suggested revisions of MIL-F-8785C, the Mil Handbook-Handling Qualities of Piloted Airplanes (Dec. 1981), attempts to rectify some of these deficiencies. The equivalent system method to be used is specified in detail and specific boundaries are placed on equivalent time delay. Existing short-period frequency or roll mode time constant boundaries are used to evaluate the equivalent parameters. This arrangement is an improvement but the lateral axis time delay boundaries are not yet defined in the document.

An alternate longitudinal requirement proposed in the handbook uses equivalent time delay and a new parameter, open-loop bandwidth, to define flying qualities boundaries. The criterion (Reference 6) uses an estimate of the equivalent time delay called τ_p; sample boundaries are shown in Figure 10. This criterion shows promise since it contains the necessary dimensions of initial delay (τ_p) and a metric (bandwidth frequency) which relates to the aircraft response.

The concept of the bandwidth criterion is in effect an open-loop version of the closed-loop Neal-Smith criterion (Reference 16). A deficiency which the criteria share is the inability to account for the sensitivity of the predicted results to small changes in the criteria constraints. This sensitivity is indeed the very system characteristic which results in "flying qualities cliffs." To be useful a criterion must expose these problem aircraft configurations which produce "explosive" flying qualities degradations with small changes in task performance standard (bandwidth) or pilot technique (compensation).

Neal-Smith Longitudinal Flying Qualities Criterion (Reference 17)

A detailed description of the genesis and evolution of this criterion is beyond the scope of this paper. Simply stated, the criterion is based on the assumption that precise pitch attitude control is essential for good flying qualities. Flying qualities boundaries were developed through correlation with in-flight simulation data which relate to the closed-loop pitch attitude task performance and the dynamic compensation necessary to achieve an appropriate closed-loop bandwidth. The criterion assumes a simple closed-loop pitch attitude tracking task and a desired bandwidth (degree of pilot task aggressiveness) which is a function of the task.

The criterion in its original form represents a very useful longitudinal design guide for the evaluation of advanced highly augmented fighter aircraft. For a variety of reasons, much effort has been expended to find alternative methods to do the same job done by the original criterion. Although the criterion is clearly not in a form to be used in a flying qualities specification, it works as well as any alternative method and is no more complex to apply than the equivalent system method. It has the advantage of being more directly related to the piloting task than other open-loop metrics. The criterion allows the complete flight control system to be evaluated without the requirement for equivalent parameter calculations. Further, evaluation of the interaction of initial delay and aircraft response is handled in one step. For the Neal-Smith criterion, the evaluation of the criterion output data is where the interpretation phase is found rather than in the initial steps as in the equivalent system process; each approach has its imperfections.

The criterion has been extended from the fighter tracking task to the precision landing task in a recent study reported in Reference 22. A version of the criterion which is applicable to the landing task was developed using the approach and landing data base of Reference 11.

In summary, the Neal-Smith criterion represents a useful longitudinal fighter flying qualities evaluation criterion for both tracking and landing tasks. As suggested in Reference 22, the criterion should be revisited and a suitable metric developed which evaluates the sensitivity of task performance to changes in bandwidth. Refinement of this closed-loop criterion approach into a form suitable for inclusion as a requirement in a flying qualities specification is likely not practical. However, recent developments in the application of the equivalent system method (preliminary MIL-F-8785C revision) have created a degree of complexity which makes the Neal-Smith approach appear more reasonable.

Development of a closed-loop time domain criterion similar to the frequency domain Neal-Smith criterion would be an appropriate research area. Systems with nonlinear features could be properly evaluated with such a criterion.

Equivalent System Criteria

As previously discussed, a longitudinal equivalent system approach has been incorporated into the latest military flying qualities specifications (Reference 21) and suggested revisions. The method specified is the frequency response method developed at the McDonnell-Douglas Company (Reference 3) and sometimes referred to as "McFit". Once the equivalent parameters are derived, they are compared with the appropriate boundaries in the specification. Generally, the same boundaries can be used that were originally developed for "classic" aircraft.

Unfortunately, this method has recently evolved into a much more complex procedure (preliminary MIL-F-8785C revision). In addition, it is surrounded with controversy related to the selection of the appropriate n_z/α value for an evaluation. However, the equivalent system method clearly has strong merit. For example, the study reported in Reference 22, evaluated the NT-33 LAHOS data base (Reference 11) using the original equivalent system method with very good results.

As noted in the discussion on MIL-F-8785C, the evaluation of the effects of control system dynamics, including pure time delay, on fighter lateral flying qualities is not well covered. The results of a recent NT-33 in-flight evaluation program (Reference 10) are of interest in this area.

This lateral higher order system (LATHOS) experiment involved very realistic tracking, refueling and formation tasks as well as precision landing tasks. A wide variety of control system effects, including time delay and prefilter lag were extensively evaluated with different levels of roll damping. Correlation of these data was obtained using a time history equivalent system approach.

The effective delay, τ_{Eff}, (to distinguish from "McFit" equivalent delay) and effective roll mode time constant, $\tau_{R_{Eff}}$, are extracted from the roll rate step response time history as illustrated in Figure 4. Finally, flying qualities boundaries on the τ_{Eff} versus $\tau_{R_{Eff}}$ plane were derived using the data for the optimum command gain. The time domain equivalent system results for the Category A tasks (tracking, refueling) are shown in Figure 11. Reasonable separation of the data is achieved. As for the longitudinal axis, a sharp degradation of pilot rating with time delay is evident. Based on these data, the allowable time delay appears to be a function of the aircraft response, in this case characterized by the effective roll mode time constant. Chalk in Reference 23 further makes the case that the allowable time delay and subsequent degradation of flying qualities is a function of the task performance standard (bandwidth) and the class of aircraft.

In summary, equivalent system methods can be used to evaluate the flying qualities of fighter aircraft with delay-prone, complex flight control systems. Flying qualities criteria based on equivalent system methods must include a definition of the method to be used.

CONCLUDING REMARKS

The flying qualities problems associated with the latest fighter aircraft are most often associated with control systems time delay. This delay is typically introduced into the initial response to a pilot input by the control system design strategies which are now achievable with today's advanced high-authority electronic control systems. In this paper we have attempted to review the meaning of time delay and its effect on fighter flying qualities. In summary, the major points in the paper are:

- Control system time delay, whatever the source, can have a profound effect on longitudinal and lateral flying qualities for precision fighter tasks.
- The allowable time delay and the rate of flying qualities degradation with time delay are a function of the level of task precision, pilot technique and the subsequent aircraft response.
- The time delay measurement method must be carefully specified and be a part of any "equivalent system" flying qualities evaluation criteria.
- Exposure of flying qualities problems related to time delay can only be accomplished with "high stress" realistic tasks.
- Although the data base is far from complete, flying qualities criteria exist which, although imperfect, can be used in the design process to avoid the flying qualities problems related to initial time delay.
- Control system designers should recognize that complexity generally results in greater initial response delay; delay-free individual elements can contribute to the overall initial delay as perceived by the pilot. Simple systems are generally better.
- More flying qualities data are required to understand fully the effects of time delay on fighter flying qualities; in particular, the effects of time delay on fighter pitch tracking flying qualities are not well defined; the effects of digital flight control system characteristics on fighter flying qualities also deserve more attention.

REFERENCES

1. Stengel, R. F., and Miller, G. E.: "Flight Tests of a Microprocessor Control System," Journal of Guidance and Control, Vol. 3, No. 6, November-December 1980.

2. [illegible]inson, J., LaManna, W. J., and Heyde, J. L.: "Handling Qualities of [illegible]ft With Stability and Control Augmentation Systems - A Fundamental [illegible]ch," Journal of Royal Aeronautical Society, February 1976.

3. [illegible]rah, R. C., Hodgkinson, J., and LaManna, W. J.: "Are Today's Specifications Appropriate for Tomorrow's Airplanes?" AGARD Flight Mechanics Panel Symposium on Stability and Control, Ottawa, Canada, September 1978.

4. Hodgkinson, J., Berger, R. L. and Bear, R. L.: "Analysis of High Order Aircraft Flight Control System Dynamics Using an Equivalent System Approach," McAIR Report 76-009, Seventh Annual Pittsburgh Conference on Modeling and Simulation, August 1976.

5. Eulrich, B. J. and Rynaski, E. G.: "Identification of Flexible Aircraft From Flight Data," AIAA Paper No. 80-1633, August 1980.

6. Hoh, R. H. and Mitchell, D. G.: "Bandwidth - A Criterion for Highly Augmented Airplanes," AIAA Paper No. 81-1890, August 1981.

7. Shafer, M. F.: "Low Order Equivalent Models of Highly Augmented Aircraft Determined From Flight Data Using Maximum Likelihood Estimation," AIAA Paper No. 80-1627, August 1980.

8. Chalk, C. R.: "Calspan Recommendations for SCR Flying Qualities Design Criteria," NASA-CR-159236, April 1980.

9. VanGool, M. F. C.: "Application of Existing Roll Response Criteria to Transport Aircraft With Advanced Flight Control Systems," AIAA Paper No. 80-1572, August 1980.

10. Monagan, S. J., Smith, R. E., and Bailey, R. E.: "Lateral-Directional Flying Qualities of Highly Augmented Fighter Aircraft," AFWAL-TR-81-3171, January 1982.

11. Smith, R. E.: "Effects of Control System Dynamics on Fighter Approach and Landing Longitudinal Flying Qualities," AFFDL-TR-78-122, March 1978.

12. Johnston, K. A. and Hodgkinson, J.: "Flying Qualities Analysis of an In-Flight Simulation of High Order Control System Effects of Fighter Aircraft Approach and Landing," McAIR Report No. MDC A5596, December 1978.

13. Smith, R. E.: "Equivalent Systems Verification and Evaluation of Augmentation Effects on Fighter Approach and Landing Flying Qualities," Calspan Report No. 6241-F-3, August 1979.

14. Bailey, R. E. and Smith, R. E.: "An In-Flight Investigation of Pilot-Induced Oscillation Suppression Filters During the Fighter Approach and Landing Task," NASA Contractor Report No. 163116, March 1982.

15. [illegible] D. T., Powers, B. G., Szalai, K. J., and Wilson, R. J.: "A Summary of [illegible]light Evaluation of Control System Pure Time Delays During Landing [illegible]he F-8 DFBW Airplane," AIAA Paper No. 80-1626, August 1980.

16. [illegible] T. P. and Smith, R. E.: "An In-Flight Investigation to Develop Control System Design Criteria for Fighter Airplanes," AFFDL-TR-70-74, December 1970.

17. Smith, R. E.: "Evaluation of F-18A, Approach and Landing Flying Qualities Using an In-Flight Simulator," Calspan Report No. 6241-F-1, February 1979.

18. Gibson, J. C.: "Flying Qualities and the Fly-By-Wire Aeroplane," AGARD Paper No. AGARD-CP-260, AGARD Conference Proceedings No. 260, Conference on Stability and Control, Neuilly Sur Seine, France.

19. Hartsfield, Col. H. W. Jr.,: "Space Shuttle Orbital Flight Testing," Society of Experimental Test Pilots, 22nd Symposium Proceedings, Technical Review, Vol. 14, September 1978.

20. Hall, G. W. and Harper, R. P.: "In-Flight Simulation of the Light Weight Fighters," AIAA Paper No. 75-986, August 1975.

21. Anon.: "Military Specification: Flying Qualities of Piloted Airplanes," MIL-F-8785C, November 1980.

22. Radford, R. C., Smith, R. E., and Bailey, R. E.: "Landing Flying Qualities Evaluation Criteria for Augmented Aircraft," NASA Contractor Report No. 163097, August 1980.

23. Weingarten, N. C. and Chalk, C. R.: "In-Flight Investigation of Large Airplane Flying Qualities for Approach and Landing," AFWAL-TR-81-3118, December 1981.

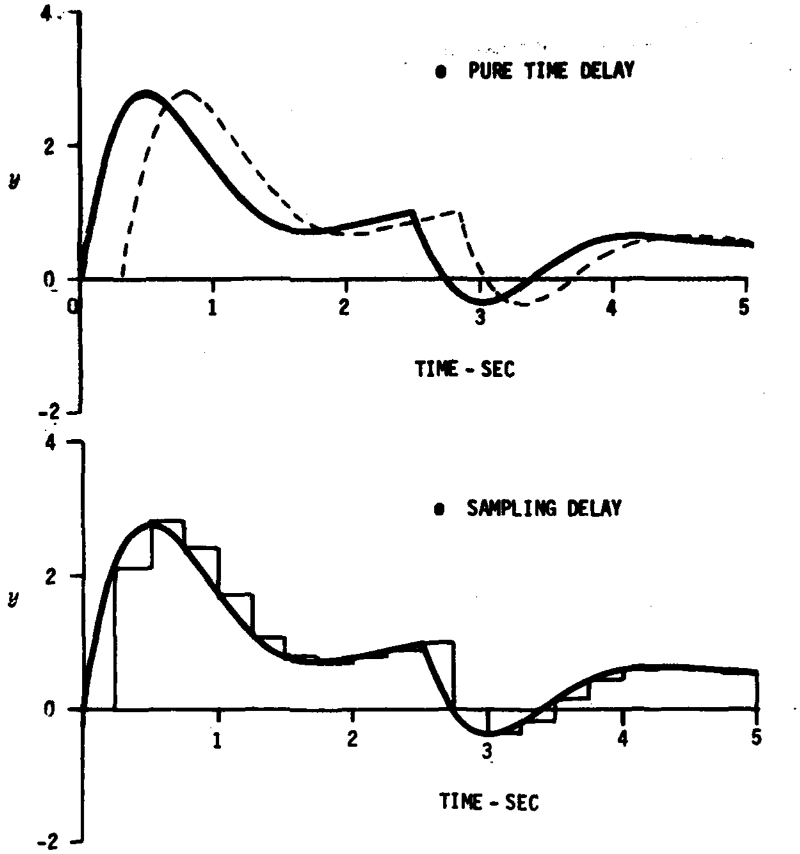


Figure 1: DIGITAL SAMPLING AND COMPUTATIONAL (PURE) TIME DELAY

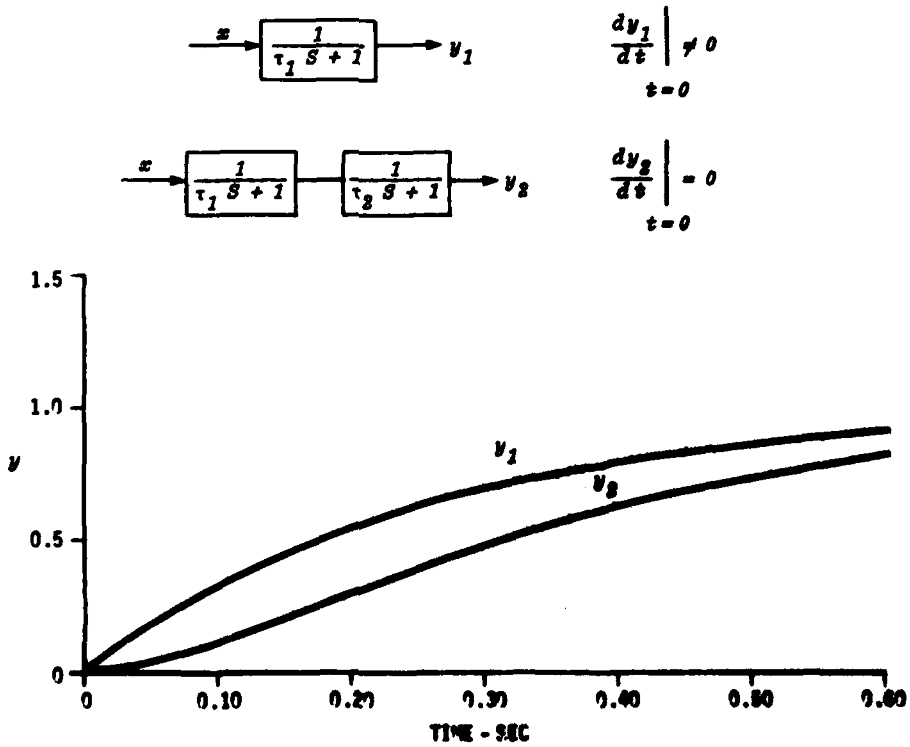


Figure 2: TIME DELAY DUE TO CASCADED FILTERS

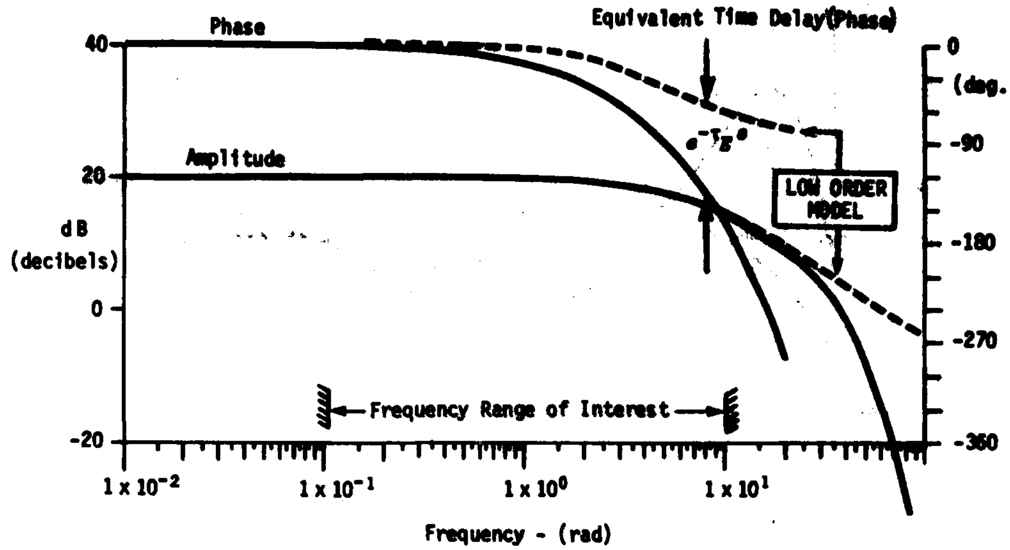


Figure 3: EQUIVALENT SYSTEM MATCH BY LOW ORDER MODEL PLUS EQUIVALENT TIME DELAY

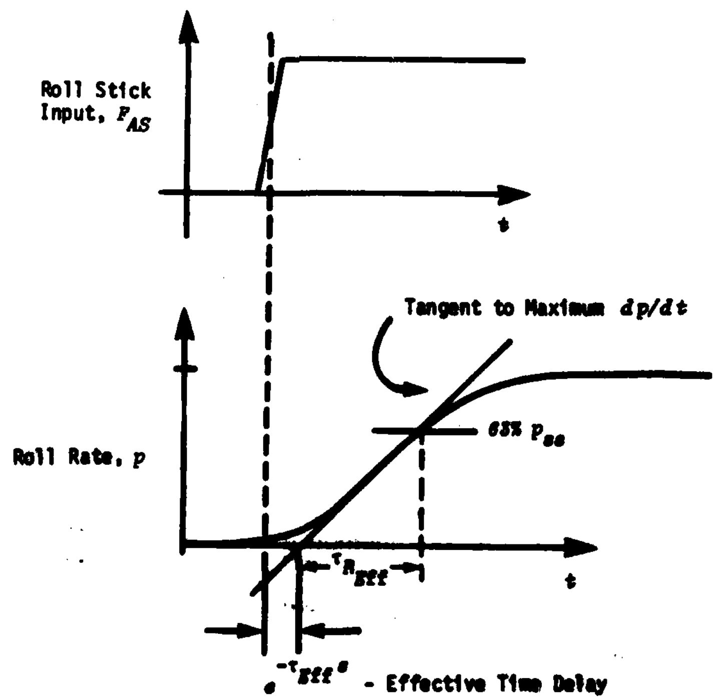


Figure 4: EFFECTIVE TIME DELAY AND TIME CONSTANT

Config. 5 - 3	τ_R	τ
Freq. Equiv.	0.16	0.051
Time Equiv.	0.16	0.045

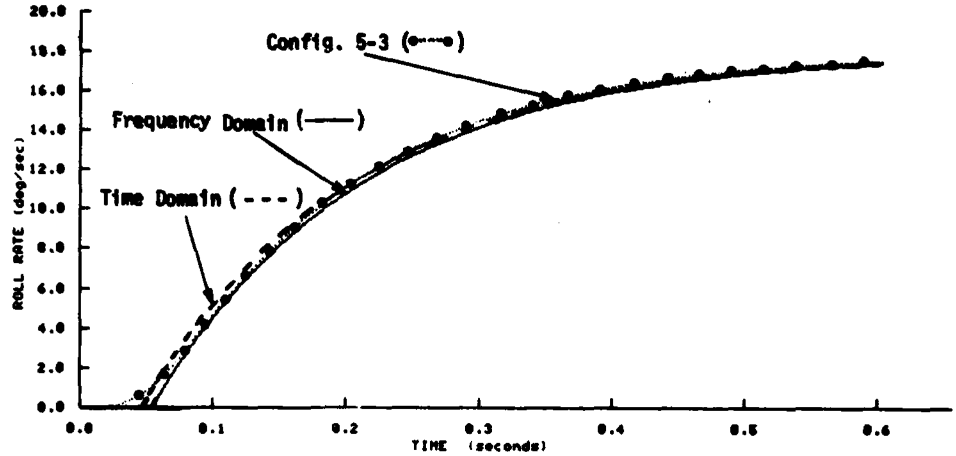


Figure 5: TIME HISTORY COMPARISON OF LOW ORDER EQUIVALENT MODELS

Config. 5-3 F1	τ_R	τ
Freq. Equiv.	0.21	0.090
Time Equiv.	0.25	0.064

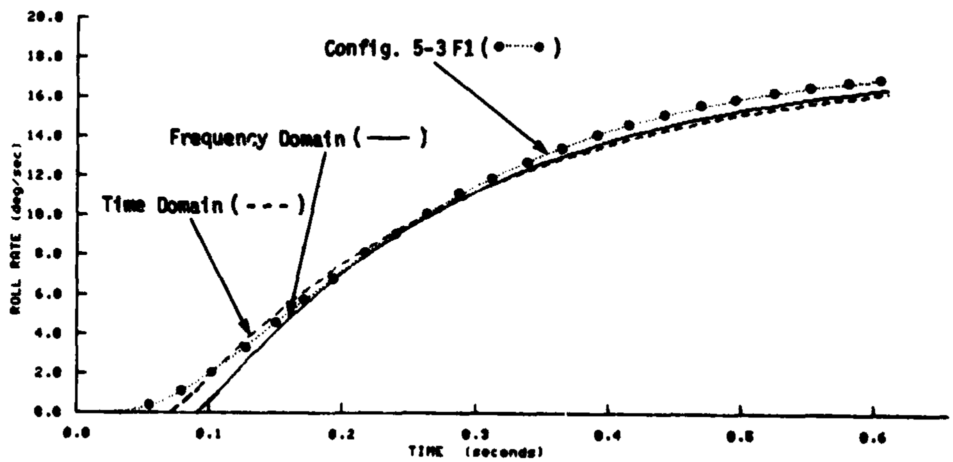


Figure 6: TIME HISTORY COMPARISON OF LOW ORDER EQUIVALENT MODELS

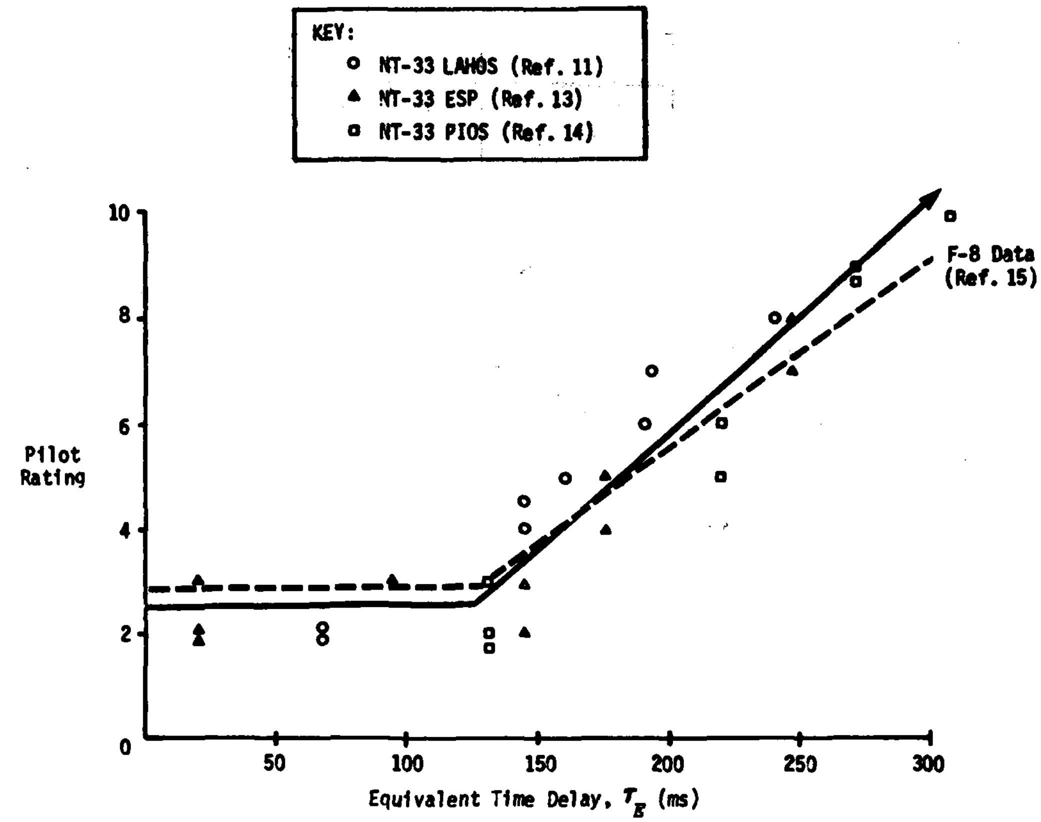


Figure 7: EFFECTS OF TIME DELAY ON FIGHTER APPROACH AND LANDING LONGITUDINAL FLYING QUALITIES

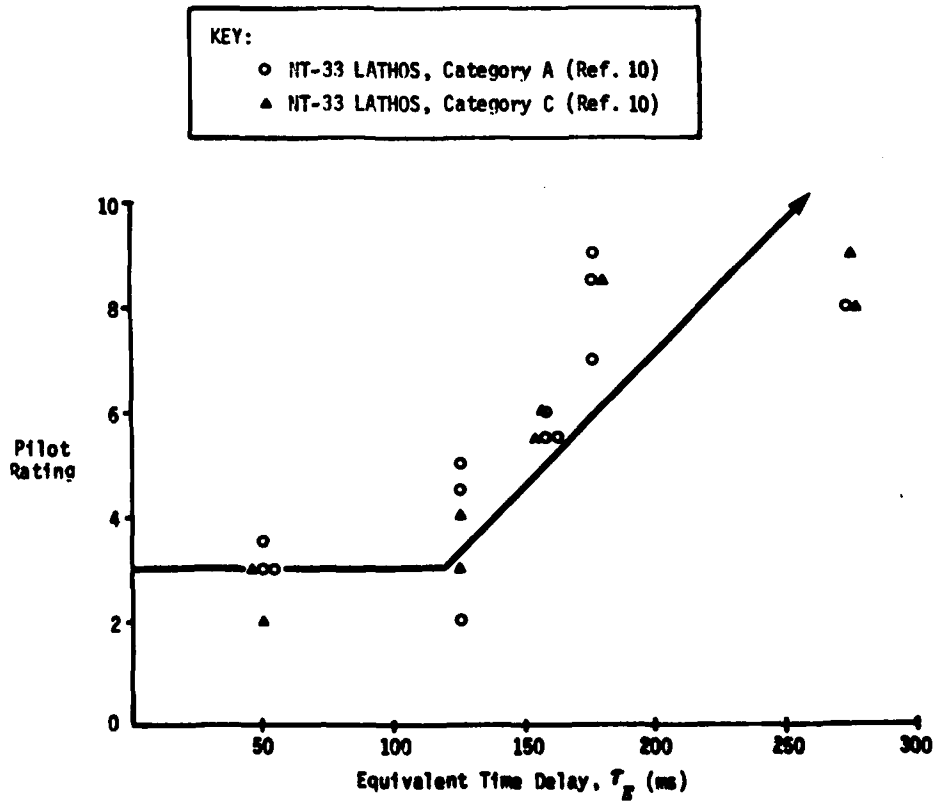


Figure 8: EFFECTS OF TIME DELAY ON FIGHTER LATERAL FLYING QUALITIES

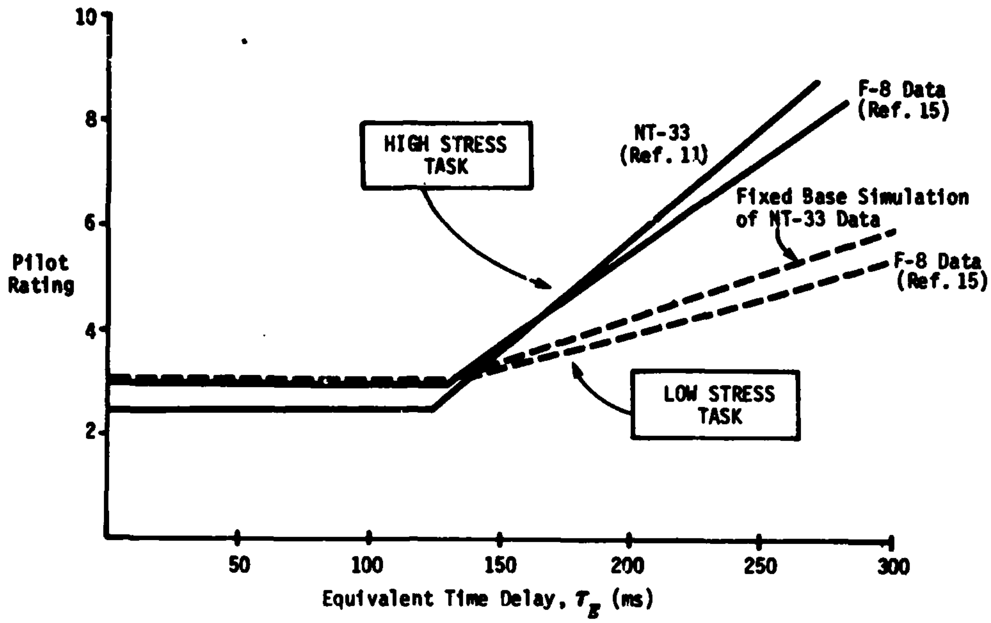


Figure 9: COMPARISON OF THE EFFECTS OF TIME DELAY FOR LOW AND HIGH STRESS LANDING TASKS

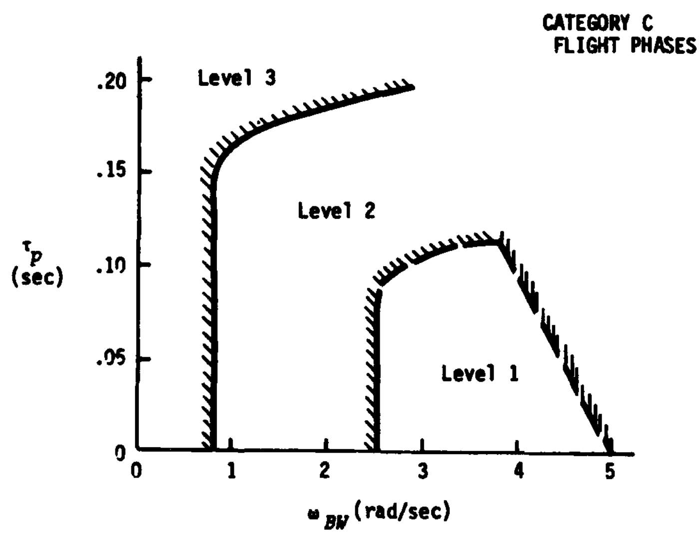


Figure 10: SUGGESTED BANDWIDTH CRITERION/TIME DELAY BOUNDARIES FOR REVISED MIL - 8785 C

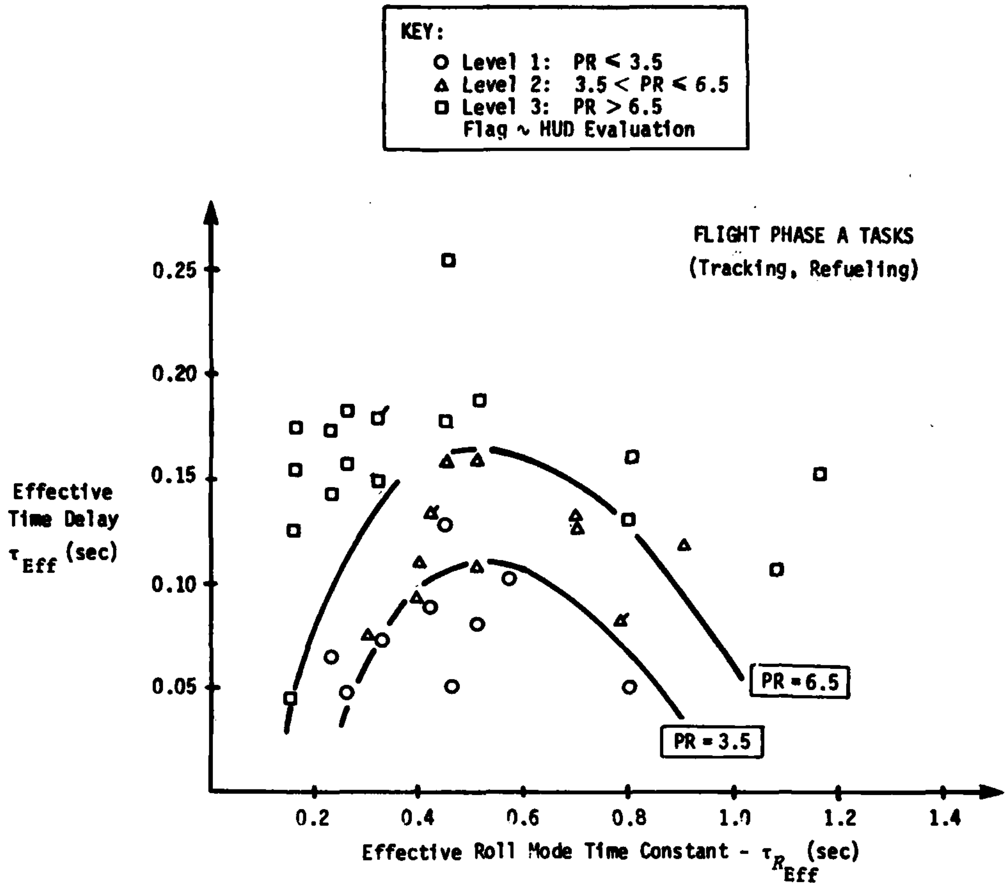


Figure 11: CORRELATION OF LATHOS DATA (REF. 10) WITH TIME-DOMAIN EQUIVALENT PARAMETERS

EFFECT OF CONTROL SYSTEM DELAYS ON FIGHTER FLYING QUALITIES

by

Captain J.T. Bakker
Royal Netherlands Air Force
Prins Clauslaan 8
2595 AJ 's-GRAVENHAGE
The Netherlands

INTRODUCTION

In most of todays fighter aircraft the pilot's stick inputs during certain flight phases are based upon information obtained from the Head Up Display (HUD). This information is generated by partly digital systems with inherent time delays, which are not included in the control system time delays described in the previous paper, although they may contribute to degradation of the flying qualities.

These comments give an example of such time delays existing in the F-16 aircraft. Apart from this some F-16 control system modifications and their resulting impact on the power approach handling qualities will be discussed.

HEAD UP DISPLAY TIME DELAYS

Under instrument conditions the pilot's stick inputs during the approach phase may be based upon information from the HUD. In the F-16 the pilot has to maintain glide path and 13 degrees Angle of Attack (AOA). HUD symbology is shown in figure 1. The flight path marker (aircraft symbol) shows aircraft velocity vector. The AOA is shown by the position of the AOA bracket with regard to the flight path marker. The flight path marker will lag (due to delayed aircraft response) attitude changes. This lag will be minimal when attitude changes are made at a slow rate. It would be incorrect to regard this lag as a time delay, because the pilot is aware of the lag and the fact that he should not consider the flight path marker as an attitude indicator.

The delay in presentation of both the flight path marker and the AOA bracket is shown in figure 2. The flight path marker information is derived from V_x, V_y and V_z transmitted in digital form from the Inertial Navigation Unit to the Fire Control Computer. Flight path marker computation takes place in the Fire Control Computer, which transmits this data to the HUD Electronic Unit. AOA information is sent from the AOA transmitter to the Central Air Data Computer. It is then transmitted in digital form to the HUD Electronic Unit, where it is combined with the flight path marker information. The combined symbology is sent to the HUD Pilot Display Unit.

Except the estimated delay of 20 milliseconds (ms) at the AOA transmitter, all other delays are sampling delays. The same assumptions as in the previous paper are used, namely that digital computational delay is not significant and that the sampling delay is one half of the sampling rate. We then find a total time delay of 30 ms for the flight path marker presentation and a delay of up to 50 ms in AOA presentation. If the pilot's stick input is made to correct an AOA deviation it will be 50 ms late already. Based upon the observations in the previous paper an additional control system time delay greater than 80 ms instead of the expected 130 ms will now degrade the flying qualities. This under the assumption that visual display system time delays have the same impact on the flying qualities as control system time delays.

CONCLUSIONS

Time delays in visual display systems may be serious enough to have an impact on flying qualities.

Time delays in such systems should be part of future investigation and, if necessary, limitations for such time delays should be included in the U.S. Military Flying Qualities Specification.

F-16 POWER APPROACH HANDLING QUALITIES

From the first days of the F-16 aircraft it is known that the aircraft possesses some undesirable handling characteristics during the approach, landing and landing roll-out. One of the many contributing factors to this undesirable behaviour are the control laws used in this flight phase. When the landing gear is lowered the control laws change from a load factor command system to a blended AOA/load factor command system. The predominant AOA command provides the pilot with a clear air speed/AOA cue. If the air speed gets slow additional stick force is required to maintain a given attitude. Some disadvantages of the system are a reduction in pitch control precision and an increased response of the aircraft to AOA distortions caused by gusts and turbulence. During the landing roll-out aerobraking is performed at 13 degrees AOA to slow down the aircraft. During this aerobraking the control system changes to ground control laws when weight on wheel switches are compressed. The air speed at which this occurs is a function of aircraft weight and centre of gravity (cg) position. It varies from 65 knows (light weight, forward cg) to 135 knots (heavy weight, aft cg). With ground control laws the AOA signal is no longer used as a control input. The transient causes an abrupt repositioning of the horizontal tail, which is noticable to the pilot because a (controllable) nose-rise will occur.

MODIFICATIONS OF THE FLIGHT CONTROL SYSTEM

Two modifications of the flight control system were made to correct problems in phases other than the approach and landing phase. The horizontal tail area was increased to reduce the possibility of a "deep stall" after a departure from the normal flight envelope. The second modification was a mechanical repositioning of the horizontal tail (rerig) to reduce the chance of a violent pitch down when electrical power to the flight control system is completely lost.

CONSEQUENCES ON POWER APPROACH HANDLING QUALITIES

The increased area of the horizontal tail causes an increase in sensitivity to pitch commands as well as an increase in tail authority in the approach configuration. This results in problems in the areas: take off, landing and landing roll-out. During take off the pilot now can raise the nose before the aircraft is ready to fly. Due to the increased pitch sensitivity there is also a risk of overrotation, which will cause the tail of the aircraft to strike the runway. The same problem exists during the landing. Touchdown is normally made at 13 degrees AOA, but abrupt stick inputs may cause overshoots big enough to strike the tail (at 15.5 degrees AOA).

Because of the increased tail authority the nose-rise during the landing roll-out at the changeover to ground control laws was much more pronounced and, especially at high aircraft weight and aft cg, sometimes not controllable. This resulted in tail strikes. The mechanical repositioning of the horizontal tail further aggrevated the abruptness and magnitude of the nose-rise because of an increase in horizontal tail travel at the above mentioned changeover.

SOLUTIONS

The solution for the nose-rise problem during landing roll-out was to increase the AOA signal fade-out time from 0.1 to 1.1 seconds. This eliminated the problem.
The problems in the take off and landing phases were less easy to solve because they required radical changes of the control laws. The blended AOA/load factor command system in the power approach configuration was replaced by a pitch rate command system. Only above 10 degrees AOA an additional AOA signal provides an AOA/air speed cue to the pilot. This AOA signal is also less dominant than the AOA signal used in the present system (Fig. 3). The modified flight control system provides more pitch attitude stability at the cost of AOA stability. Below 10 degrees AOA the aircraft is neutrally AOA stable, up to 14 degrees AOA only a small amount of stick force is required to increase AOA and above 14 degrees AOA the stick force required to increase AOA is raised with a factor 3. This results in a better pitch response of the aircraft, which reduces the tendency to overcontrol. Additional benefits are that the aircraft is less susceptible to gusts and turbulence, also undesirable pitch transients during landing gear selection are no longer present.

Due to the various Air Forces using the F-16, numerous test pilots were involved in the evaluation of the modified control system. Final test flights have been completed and the decision to incorporate the modified control system in the F-16 will be taken soon. Incorporation will be subject to high cost due to the required hardware changes in the analog flight control system and the number of aircraft already in service.

CONCLUSION

Any improvement to a flight control system will almost certainly create problems in another area, therefore even small deficiencies in control systems should be corrected at the earliest possible stage.

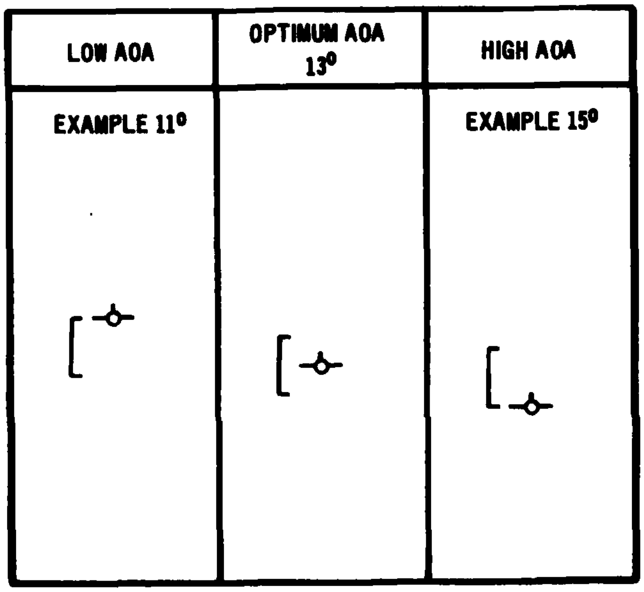


Fig. 1 HUD angle of attack symbology

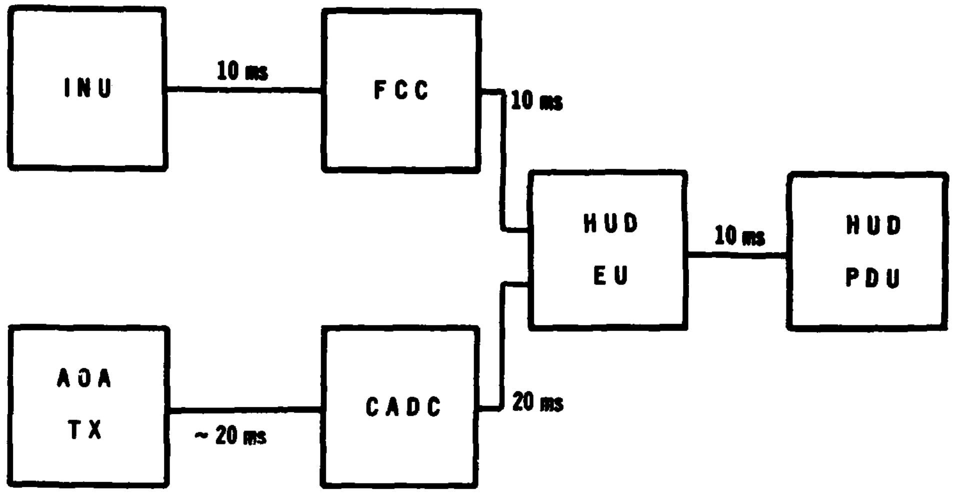


Fig. 2 Flight path marker and AOA time delays

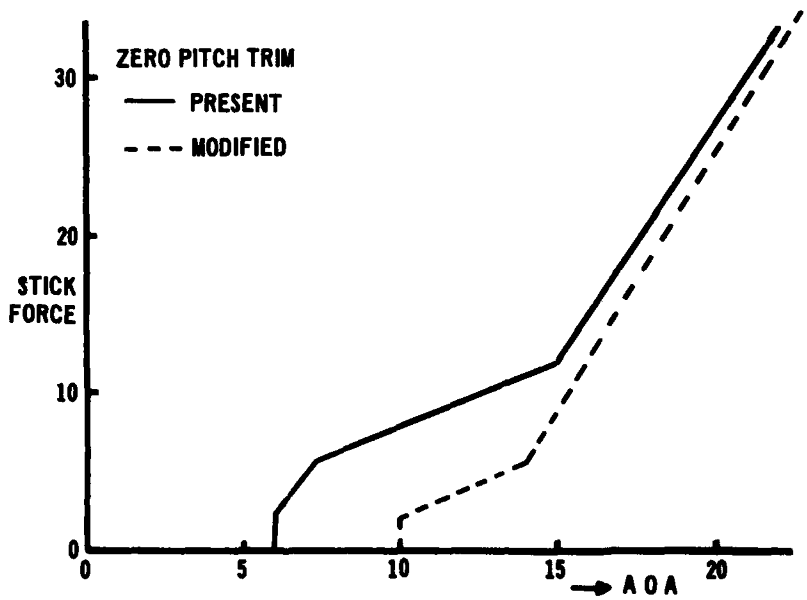


Fig. 3 Speed stability of present and modified system

AN EXAMPLE OF LONGITUDINAL AND TRANSVERSAL OSCILLATION COUPLING :
THE EPSILON AIRCRAFT "CORK SCREW"

Author : J. IRVOAS
AEROSPATIALE
31053 TOULOUSE - FRANCE

SUMMARY

Few of the rules an aircraft manufacturer has to respect to meet the regulation flight quality criteria warn him against possible bad interactions which may exist between the classical oscillations of an aircraft. Generally Dutch roll and incidence oscillation are modes which are so separate that the criteria attributed to each of them are specific and their analysis at the time of the aircraft design is performed separately by uncoupling "lateral" and "longitudinal" equation. Hence the surprise when a prototype military trainer aircraft manufactured by Aerospatiale "the EPSILON", revealed, during its 1st flights in spring 1980, a sustained oscillation which for want of correct name, in the conventional vocabulary of the best authorities on flight mechanics, was called "cork screw". All instrumentations, whatever the orientation of their sensitive axes, bore the sign of this combined movement of incidence oscillation and Dutch roll.

This was the starting point of an analysis which required implementation of the full panoply available to the engineer : wind tunnel tests on models installed on a "yaw-pitch" head simulation of the phenomenon by modeling with six degrees of freedom.

An incidence oscillation and yawing combination criterion, the search for influent aircraft architecturer and the implementation of a solution succesfully tested in flight on the preproduction aircraft resulted from this analysis.

NOTATIONS

Cm_β	Gradient of aerodynamic pitching moment coefficient with sideslip
β	Sideslip
Cn_α	Gradient of aerodynamic yaw moment coefficient with incidence
α	Incidence
ρ	Specific weight of the air
S	Aircraft wing reference area
l	Aircraft wing reference chord
m	Aircraft weight
ρ_y	Radius of giration in pitch
ρ_z	Radius of giration in yaw
C_Z	Gradient of aerodynamic lift coefficient with incidence
Cmq	Gradient of aerodynamic pitching moment coefficient with reduced pitch angular velocity $q\frac{l}{V}$
V	Aircraft speed
Cy_β	Gradient of aerodynamic lateral lift coefficient with sideslip
Cn_r	Gradient of aerodynamic yawing moment coefficient with reduced yaw angular velocity $r\frac{l}{V}$
Cn_β	Gradient of aerodynamic yawing moment coefficient with sideslip
r	Yaw angular velocity
q	Pitch angular velocity
Sd	Fin area
ld	Distance from the aerodynamic centre of the fin to the aerodynamic centre of the wing.

PLAN

1. Introduction
2. The phenomenon on the prototype aircraft
3. Intuitive explanation of the phenomenon
4. Mechanical explanation : modeling of the phenomenon
5. Physical explanation
6. The solution : the production aircraft

1. INTRODUCTION

The tail cone design of the EPSILON, a two-seater economic initial trainer aircraft designed for the French Air Force, presented a very particular problem.

This aircraft is characterized by a high power ratio provided by a single 300 H-P AVCO LYCOMING propeller engine, allowing the aircraft to fly at 200 kts.

The prototype presented a Dutch roll greatly disturbed by the slipstream of such a propeller. This led Aerospatiale to study the interaction of this slipstream with the aerodynamics of the aircraft with the greatest care. This study resulted in the preproduction aircraft design. This report deals with this experience.

2. THE PHENOMENON ENCOUNTERED ON THE PROTOTYPE AIRCRAFT

The phenomenon occurred from the very 1st flights : either during stabilized yawing tests or during actuation of the rudder to identify the lateral characteristics of the aircraft.

Whereas the Dutch roll converged slowly but steadily as expected on some occasions (sheet 2), the convergence presented variations during other flights (sheet 3). Sometimes the divergence (sheet 4) was so large that the yawed flight could not be stabilized.

The defect mainly appeared in clean configuration on right sideslip. The phenomenon appeared as a "cork screw" movement to the pilot (sheet 5) : it was a combined "yaw-pitch" oscillation close to the quadrature, the aircraft nose describing an almost circular anti-clockwise movement.

The amplitude of the phenomenon could reach a ± 2° sideslip at 200 kts, associated with a ± 1.25° incidence.

With flaps and slats extended, the defect appeared less clearly with a strong left sideslip, the rotation was then clockwise.

3. INTUITIVE EXPLANATION OF THE PHENOMENON

Of course, the phenomenon can easily be understood when one is aware of the "aerodynamic facts" which occur. It is nevertheless necessary to link them to reach a logical reasoning : a vicious circle has to be built, since we were dealing with a self-sustained movement. We are going to understand intuitively that a combination of 2 aerodynamic facts is necessary. The first fact is an effect of the incidence on the yawing-torque (sheet 6). Let us consider that the aircraft is side-slipping to the right, the wind vane effect (return torque) being countered by an opposite rudder deflection (stabilized side slip).

Let us imagine a disturbance which makes the aircraft pitch-up (increased incidence). A loss of the return torque is noted, hence an increase in sideslip.

Let us continue immediately with the 2nd fact : an increase in sideslip causes a nose-down torque (sheet 7). The increase in sideslip is therefore immediately followed by a decrease in incidence. At this point in our explanation, the nose of the aircraft has described a semi-circle since its initial pitch-up disturbance. The lower semi-circle can be explained in the same way from the nose down movement in which we left our aircraft. The circle thus closes, and we can imagine that under some favourable conditions the movement manager to become self sustained.

At this point in our "explanation", 2 questions avise for the enginer who wants to understand :
- what conditions are to be fulfilled for the movement to become self-sustained ?
- what causes the aerodynamic facts presented ?

We will try to answer these questions in the following paragraphs. The aerodynamics characteristics involved in this phonomenon $Cm(\alpha, \beta)$ and $Cn(\alpha, \beta)$, have been revealed on a model in the wind-tunnel. In chapter 5 we will describe the important wind-tunnel test campaign which was carried out in order to understand and cure the phenomenon.

4. MECHANICAL EXPLANATION : MODELING OF THE PHENOMENON

It is easy to answer the 1st question : what conditions are to be fulfilled for the movement to become self-sustained ? Flight mechanics help us to do so.

Aircraft "modes" are usually analysed from its equations linearized and uncoupled into two systems : a longitudinal one, and a lateral one. A first approximation of the Dutch roll and the incidence oscillation is thus easily defined (sheet 8).

Linearization and uncoupling : two operations still used, but which will probably be given up some day and be replaced by the bifurcation methods (ref. 1) based on the catastrophe theory. As we cannot present our phenomenon using this method (l'Office National de Recherches en Aeronautique -ONERA- is concentrating on it), we will go on linearizing.

On the other hand, we shall not uncouple, since the phenomenon we are interested in can precisely be explained by a coupling between the longitudinal and lateral system. The coupling appears through excitation of the incidence oscillation by the sideslip (Cm_β) and an excitation of the Dutch roll by the incidence (Cn_α). As shown on Nyquist's drawing, the combination of these two couplings can cause a divergent mode in the following necessary, but not sufficient condition :

$$Cn_\alpha \cdot Cm_\beta > 0$$

Sideslip and incidence are then in quadrature. Nyquist's drawing on sheet 8 illustrates the case encountered in cruise around which Cn_α and Cm_β are both negative. The incidence is a phase ahead of the sideslip : the rotation of the aircraft nose, seen by the pilot, is indeed anti clockwise.

The aircraft modes, taking longitudinal-lateral coupling into account, are given by the characteristic equation of the full system of linearized equation of the aircraft (sheet 9). For the sake of clarity we have neglected the roll equation here : this is perfectly acceptable, because this simplification is generally justified at high aircraft speeds.

Uncoupled system modes, as we can see, are but a first approximation of the real modes : a 1st approximation that is sufficient in the case of the incidence oscillation, but rough in the case of Dutch roll in sofar as its proximity to the imaginary axis may well give rise to a divergence.

The sufficient condition for it to appear is thus expressed (sheet 10) :

$$Cn_\alpha . Cm_\beta > \left(\frac{\rho S \ell}{2m}\right)\left(\frac{\rho_y}{\ell}\right)^2 \left[C_{z\alpha} + \left(\frac{\ell}{\rho_y}\right)^2 C_{mq}\right]\left[C_{y\beta} + \left(\frac{\ell}{\rho_y}\right)^2 C_{nr}\right] Cm_\beta$$

What was the situation for our aircraft with respect to this criterion ? (sheet 11). As we can see, all the conditions to cause a divergence are present ! the divergences met in flight were reproduced in simulation (sheet 12).

It was the 1st time such a phonomenon occurred on aircraft manufactured by us. We therefore gave in to our curiosity and sought to find out how they stood with respect to this criterion (sheet 13).

This table reveals sensitivity to any latent coupling of our EPSILON prototype. The low value of the product Cn_r Cn_β of this aircraft is striking. Let us analyse the physical causes of this sensitivity.

5. PHYSICAL EXPLANATION

The sensitivy of an aircraft to the phenomenon previously described depends on the respective levels of the product $Cn_\beta . Cn_r$ on the one hand and of the product $Cn_\alpha \cdot Cm_\beta$ on the other.

5.1 Value of the coefficients $Cn\beta$ et Cnr

The low values of $Cn\beta$ and of Cnr are always associated with a small fin area (Sd) or a short fuselage (ld) : this is shown clearly on sheet 13. The low values of $Sd.ld$ are characteristic of a single engine aircraft : this type of aircraft does not need a yaw torque to counter the failure of an off-centered engine.

We think therefore that these simple engine aircraft are more likely to present our phenomenon.

This low value of coefficients Cnr and $Cn\beta$ however is not disturbing insofar as the coefficients $Cn\alpha$ and $Cm\beta$ are small. This is what we managed to achieve on the production aircraft.

Let us try to see why these crossed coefficients were important on the prototype aircraft.

5.2 Value of coefficients $Cn\alpha$ and $Cm\beta$

a) Influence of the incidence on the yaw torque ($Cn\alpha$)

The numerous wind tunnel tests have been carried out on very different configurations of the model :
- with and without engine (airframe),
- with and without horizontal stabilizer,
- for various tail-cone configurations,
- for various sideslip and various incidences.

These tests have revealed that the airframe presents a wind vane effect the gradient ($Cn\beta$) of which decreases under the effect of incidence : at strong sideslips (stabilized sideslips) the yaw return torque due to sideslip is thus already very sensitive to the incidence. The addition of engines on the model tends to extend the effect of the incidence and make it more uniform in a wide range of positive sideslip (sheet 14).

This effect is naturally associated with the direction of rotation of the propeller. We shall notice that the addition of an engine has an important effect only where the aircraft incidence increases.

The tail cone modifications we will deal with in the next chapter have reduced only slightly this gradient $Cn\alpha$.

b) Influence of sideslip on the pitching moment ($Cm\beta$)

We have noted in the wind tunnel that the pitch-down moment due to sideslip already existed on a model with no engine as well as on a model without horizontal stabilizer.

We notice this effect on all aircraft with low wings (EPSILON, A300). On the contrary, we do not meet it on the model of our high wing aircraft project : the ATR 42 (commuter aircraft). The initial characteristics of the EPSILON or ATR are not modified by their models being equipped with a horizontal stabilizer at the middle or top of the fin : the sideslip remains neutral on the pitching moment of the ATR ; it remains pitch down on the EPSILON prototype.

On the contrary, equipping the EPSILON and ATR models with a low stabilizer, corrects the $Cm\beta$ effect : the sideslip pitch-down effect is hardly perceptible on the EPSILON (sheet 15) ; the sideslip becomes even pitch-up on the ATR (sheet 16).

6. THE SOLUTION : THE PRODUCTION AIRCRAFT

Considering what has just been said, one will understand our line of action to cure this "cock screw" phenomenon.

Action took place on two fronts :
- the fin efficiency has been increased : new area, new aspect ratio (see sheet 17),
- the horizontal stabilizer has been lowered, both because of its position on the tail cone, and of the new design of the tail cone itself.

Wind tunnel tests of such a configuration have revealed a 25 % increase in coefficients $Cn\beta$ and Cn_r on the are hand, and a very important decrease in the coupling term product $Cn\alpha . Cm\beta$, on the other (see sheet 18).

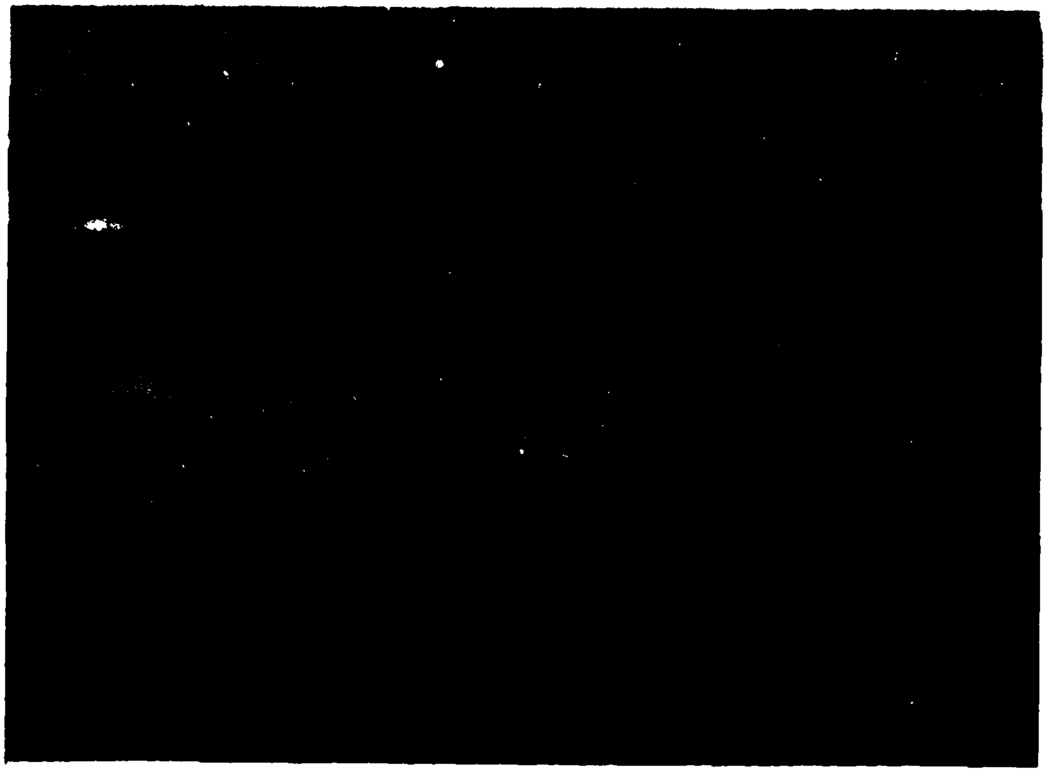
All these modifications have brought the aircraft bach to an acceptable stability range, presenting a reassuring safety margin as indicated by the criterion we have defined.

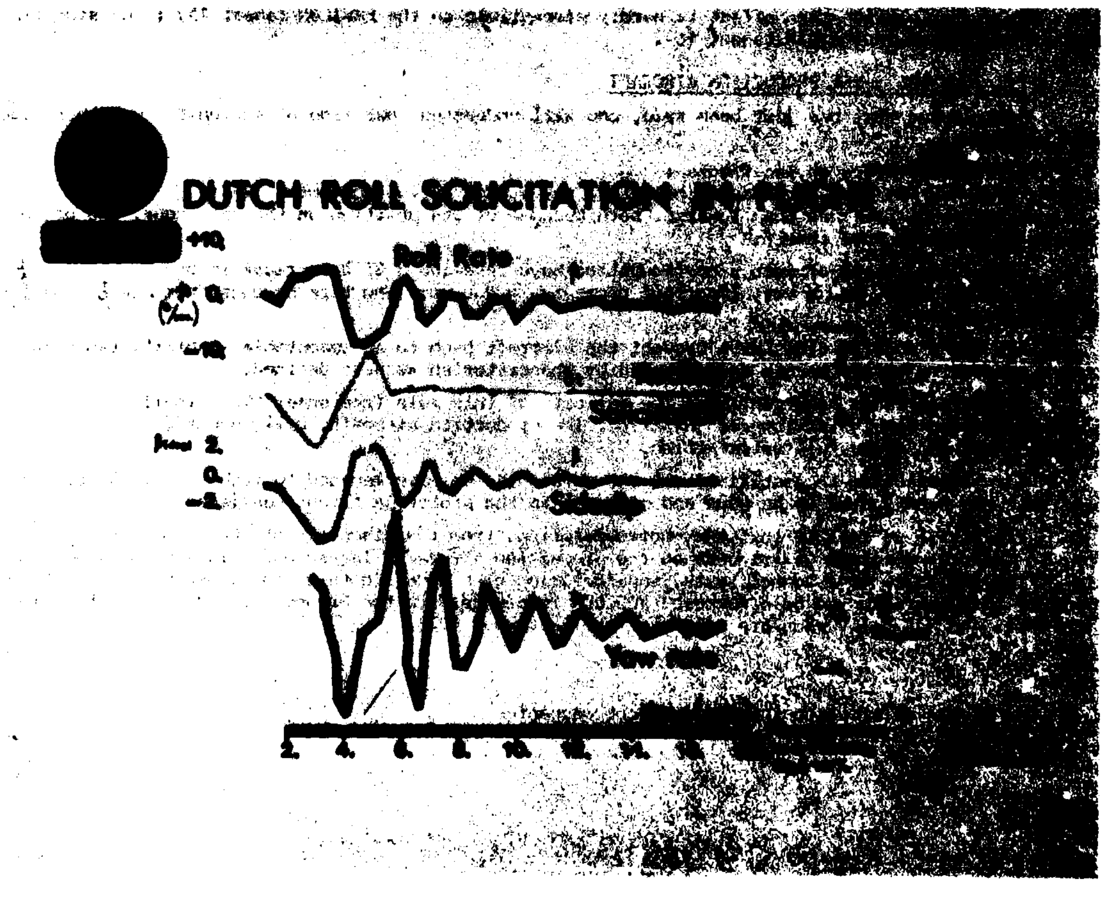
Simulation tests have confirmed the validity of this rule (see sheet 19). Finally, test flights have proved that our line of action was justified. The production aircraft, built according to these precepts (see sheet 20) gives full satisfaction.

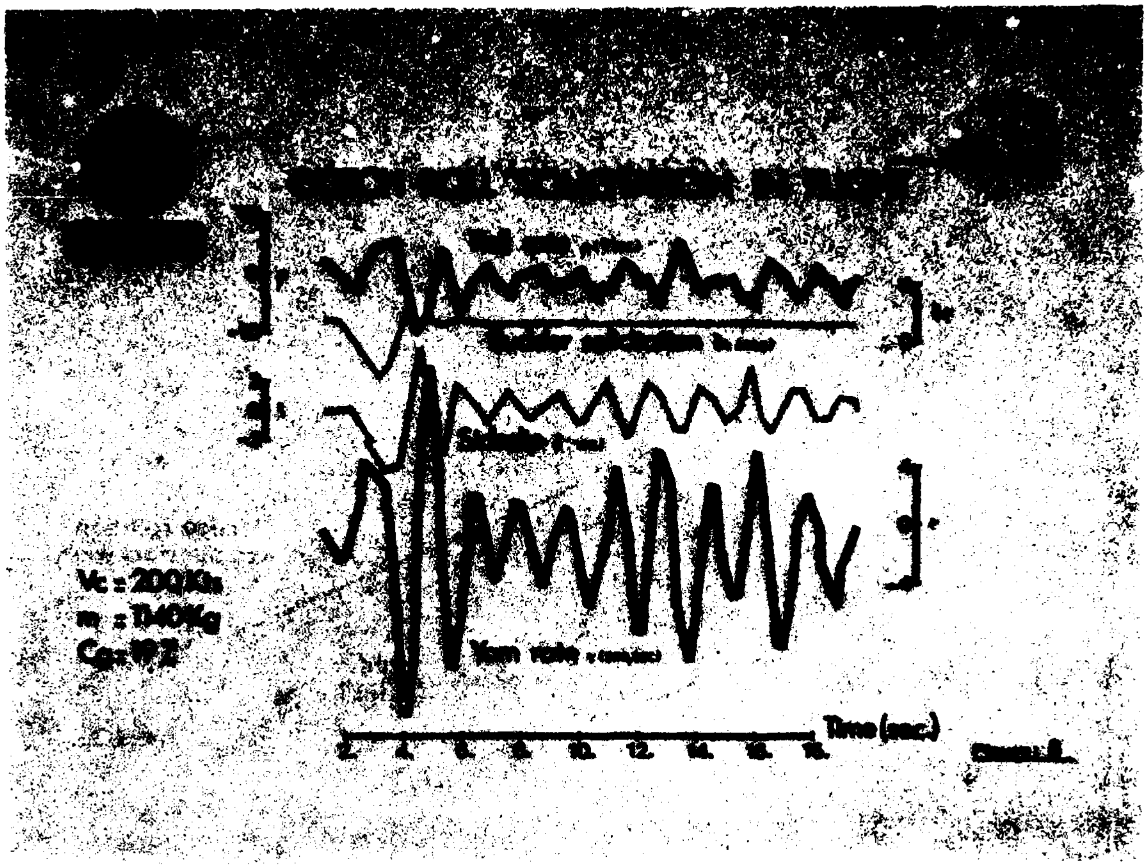
Our satisfaction is justified, all the more so as we have managed to resolve this problem while keeping what had been judged to be good and pleasant in the prototype flight qualities.

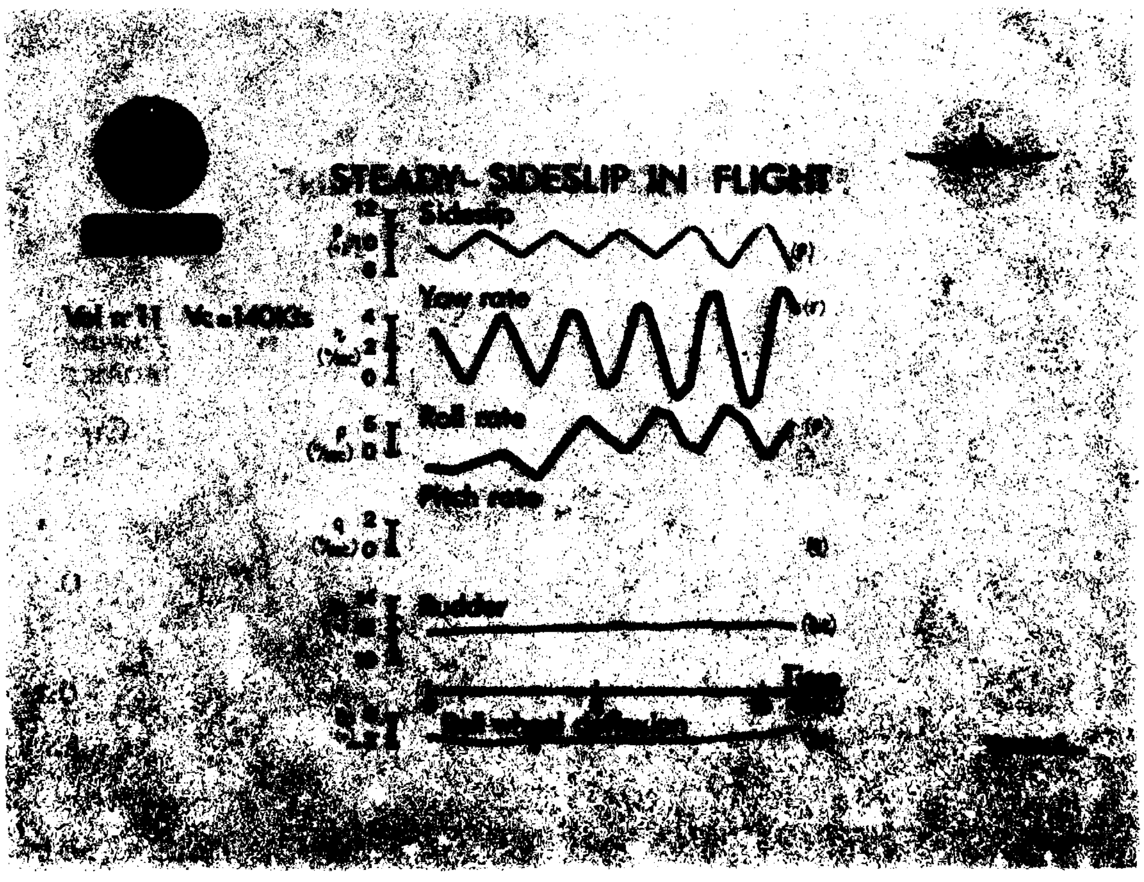
Finally if we realize that the whole operation, from the discovery of the phenomenon on the prototype, to the demonstration of its cure on the production aircraft, lasted only six months (including Design Office analysis, wind tunnel tests, manufacturing of a modofied tail cone), one will understand why we condider this operation to be a success. EPSILON in now sold to the French Air Force which has ordered 150 aircraft. The 1st deliveries are scheduled for September 1983.

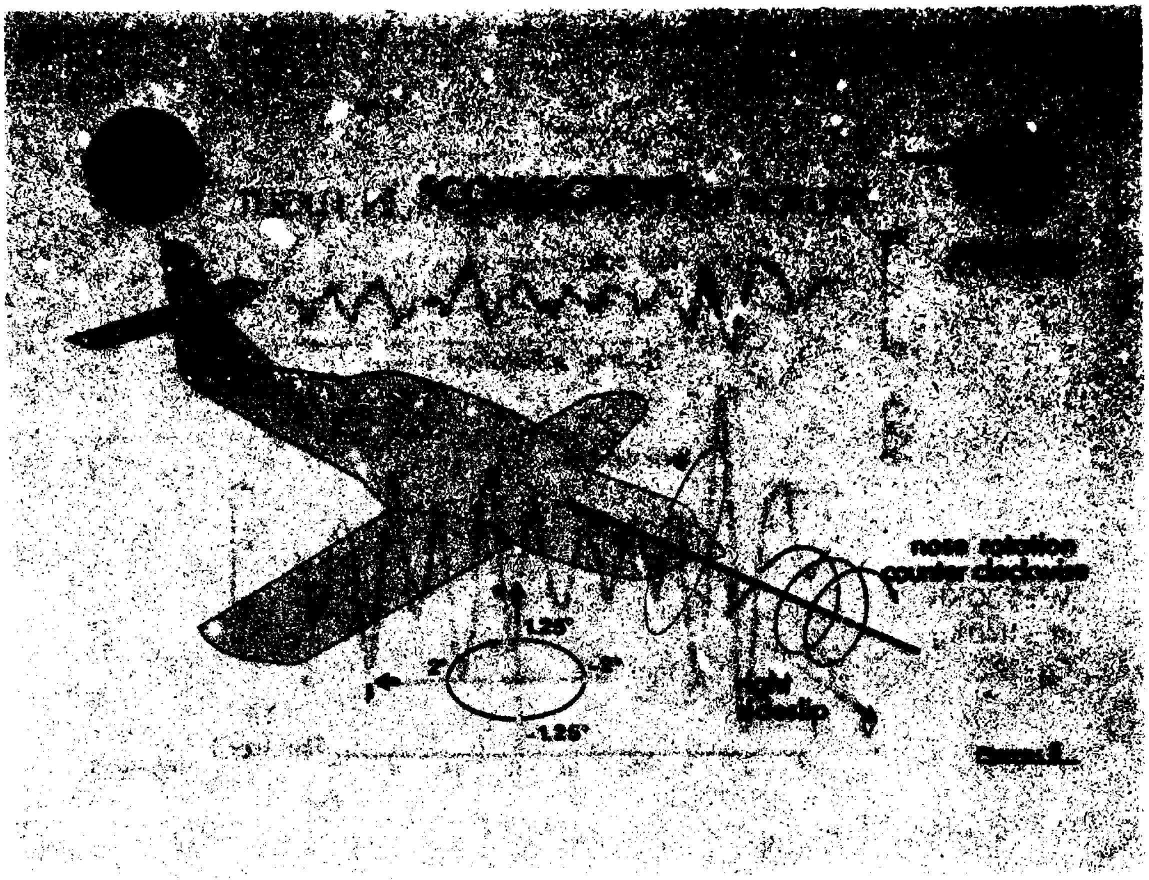
Ref. 1 : Use of the bifurcation theory in flight mechanics
ONERA - P. GUICHETEAU - Novembre 1979 - ref. RT 2/7225 SY.

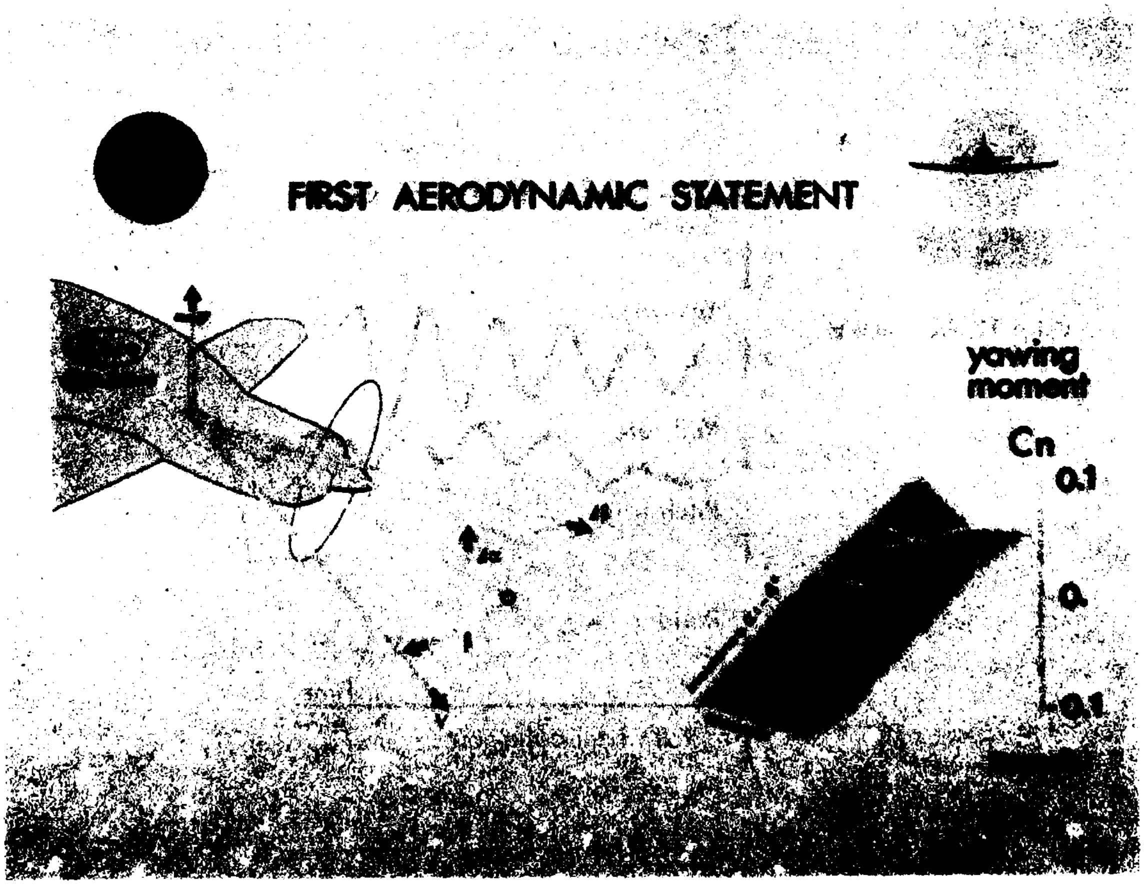


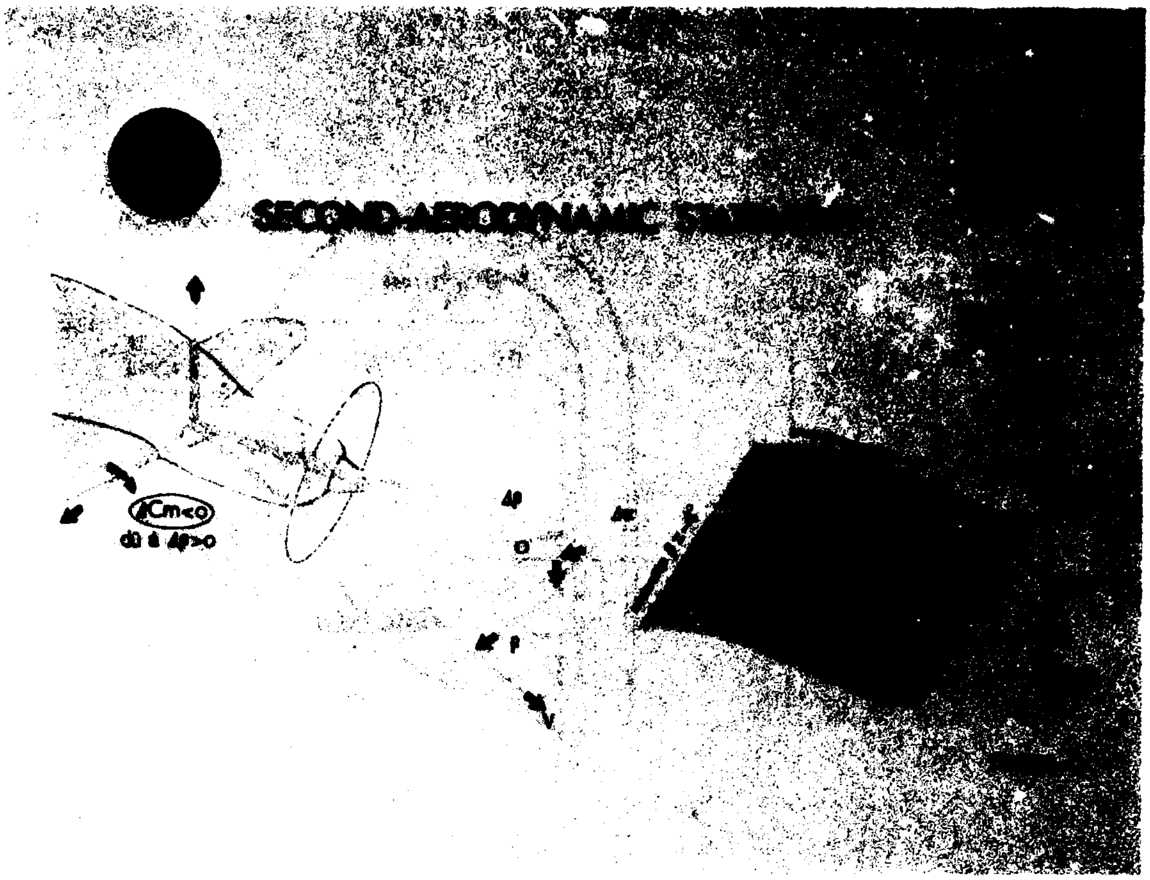

DUTCH ROLL SOLICITATION
Roll Rate
+10.
0.
-10.
2.
0.
-2.
Yaw Rate
2.
4.
6.
8.
10.

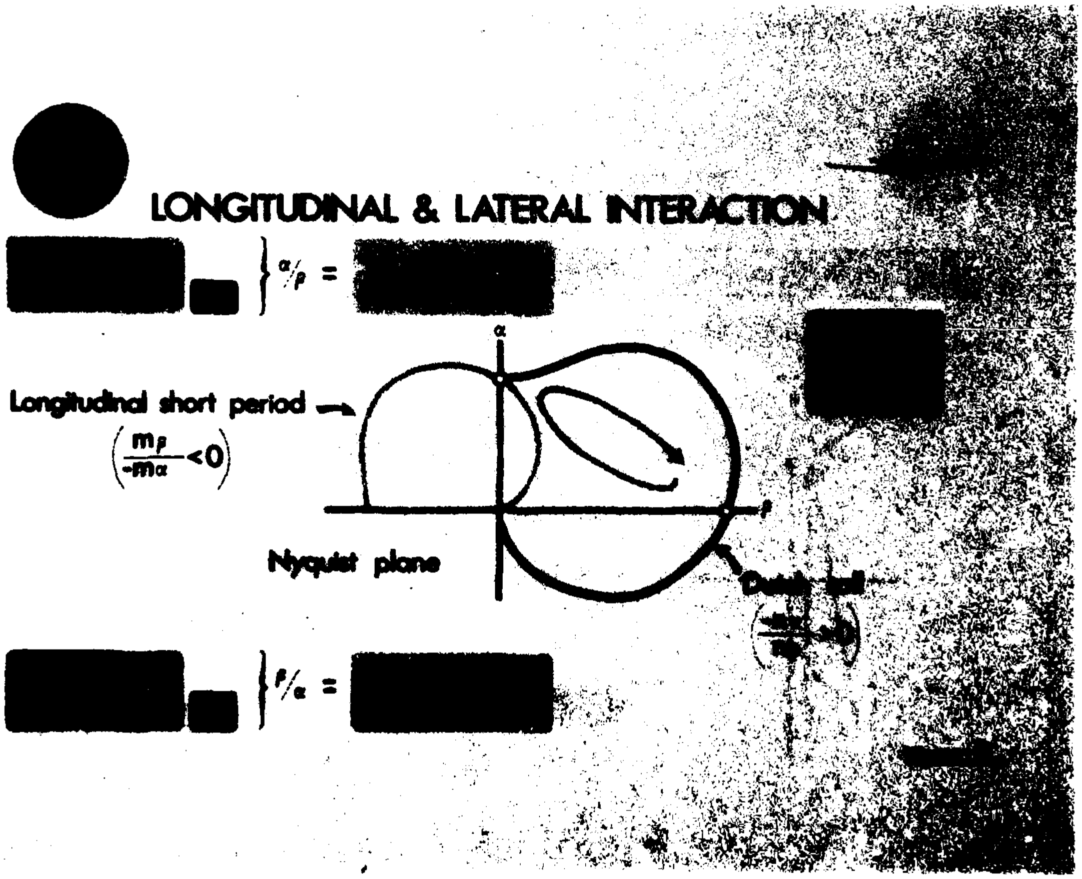

Sideslip
Yaw rate
Vc = 200Kts
m = 1140Kg
Time (sec.)
2.
4.
6.
8.
10.
12.
14.
16.
18.

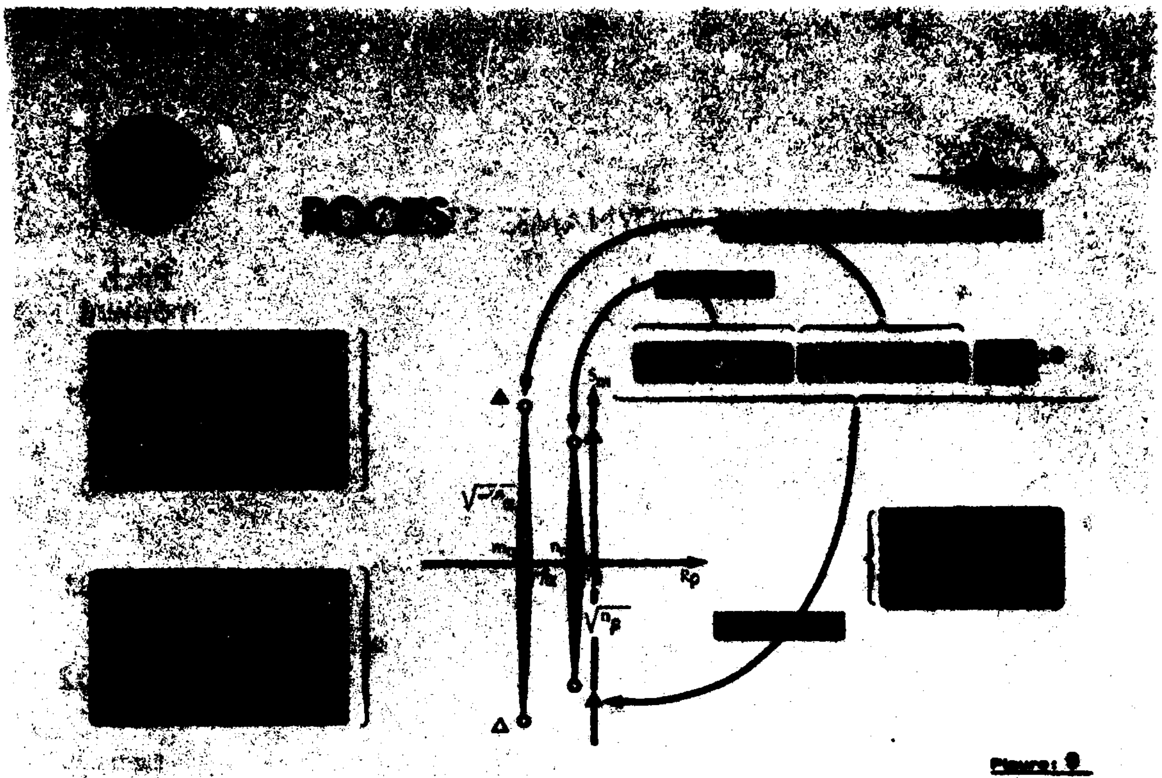

STEADY-SIDESLIP IN FLIGHT
Sideslip
Vc = 140Kts
Yaw rate
Roll rate
Pitch rate
Rudder
Time

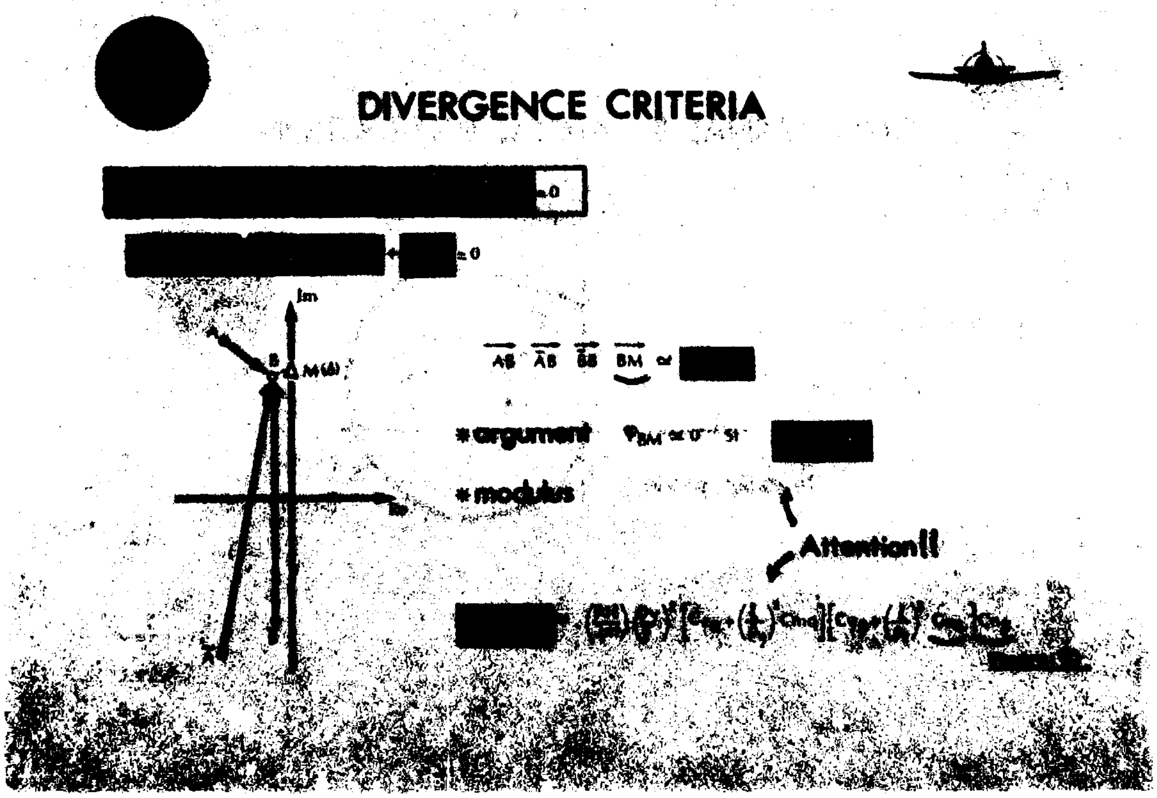

nose rotation
counter clockwise
1.25°
-1.25°


FIRST AERODYNAMIC STATEMENT
yawing
moment
Cn
0.1
0.




LONGITUDINAL & LATERAL INTERACTION
α
Longitudinal short period
Nyquist plane


[redacted]
$\sqrt{n_\beta}$
Figure: 9


DIVERGENCE CRITERIA
[redacted] = 0
[redacted] = 0
Im
B
M(δ)
AB AB BB BM ∝ [redacted]
* argument [redacted]
* modulus
Attention!!
[redacted]

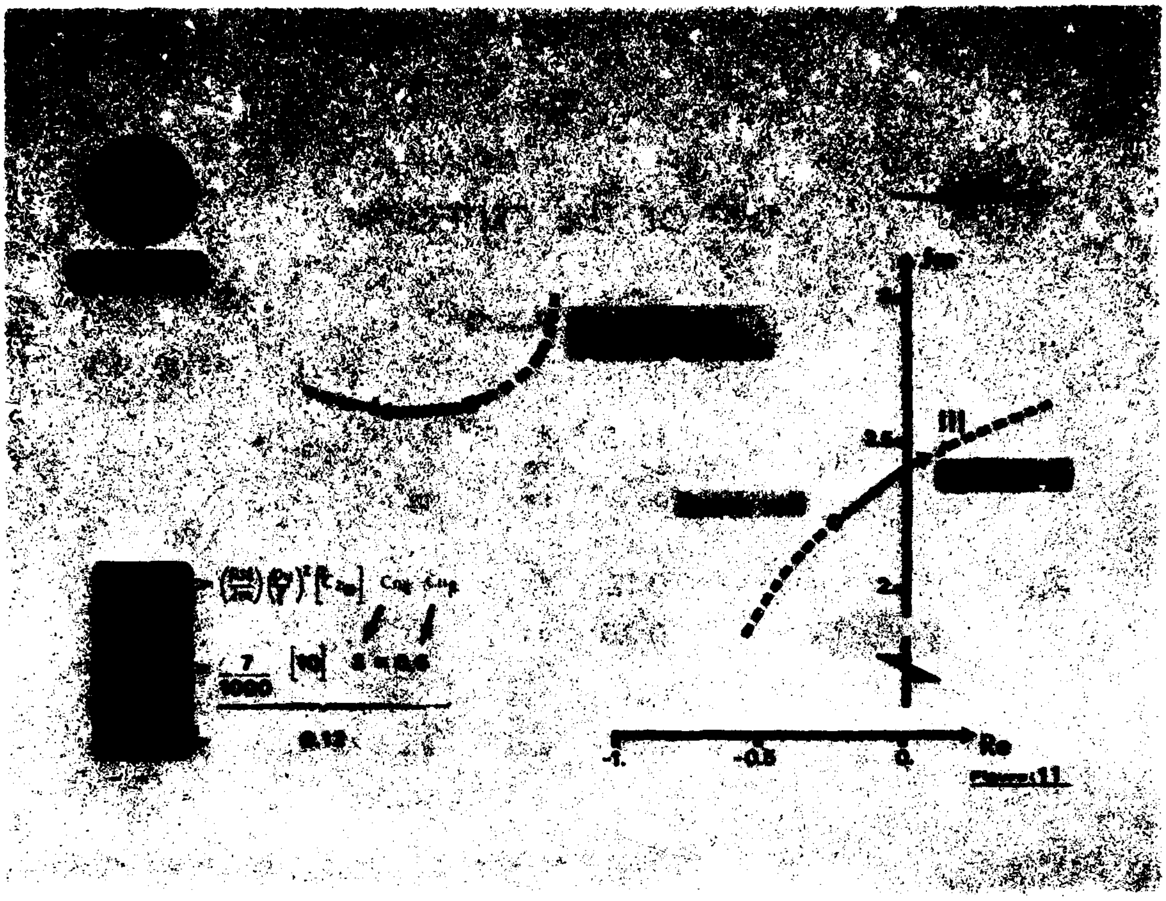


Figure 11

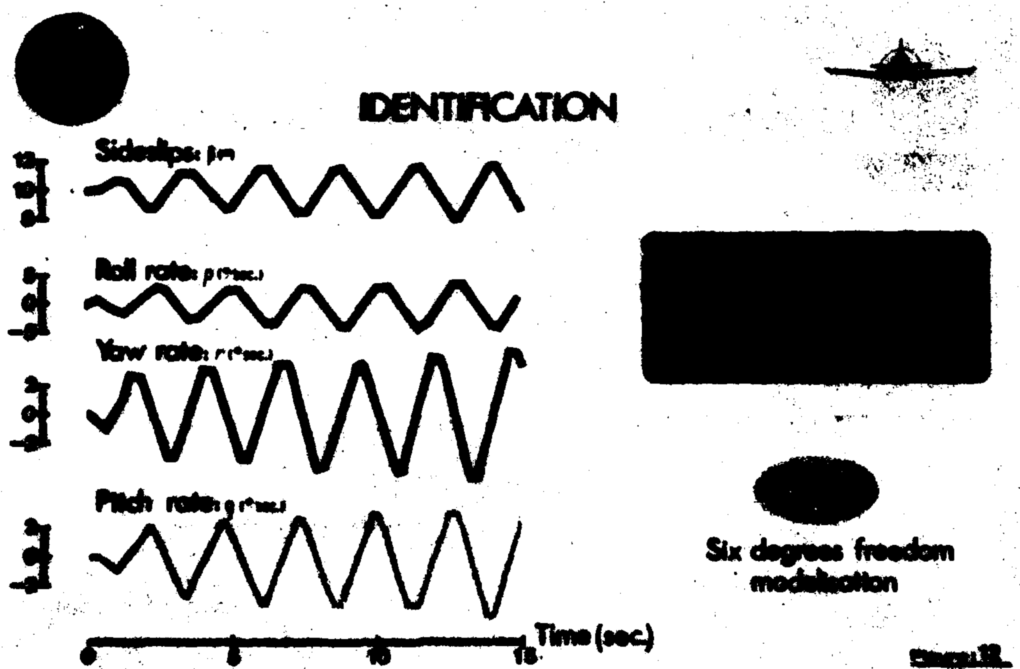


Figure 12

TEST OF THE CRITERION

A300 ATR42

$$\frac{\rho s \ell}{2m}\left(\frac{\rho v}{\ell}\right)^2 \left[C_{z\alpha} + \left(\frac{\ell}{\rho v}\right)^2 C_{mq}\right]\left[C_{y\beta} + \left(\frac{\ell}{\rho r}\right)^2 C_{nr}\right] C_{n\beta}$$

4 10^{-3}	3 10^{-3}	3 10^{-3}	$\frac{\rho s \ell}{2m}$
3	7.5	9	$-C_{nr}$
0.6	1.6	1.2	$C_{n\beta}$
0.12	0.5	0.5	
0.25	0.1	0.1	
0.44	0.62	0.30	$\frac{s_d}{s}\frac{\ell_d}{\ell}$
1.3	2.2	3.6	$\frac{s_d}{s}\left(\frac{\ell_d}{\ell}\right)^2$

Figure: 13

INCIDENCE EFFECT ON THE YAWING MOMENT

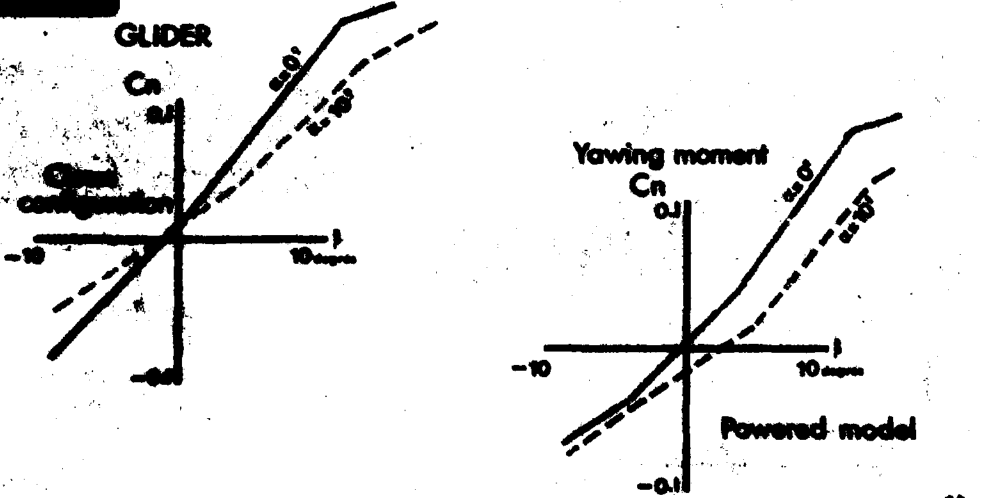


Figure: 14

SIDESLIP EFFECT ON THE PITCHING MOMENT

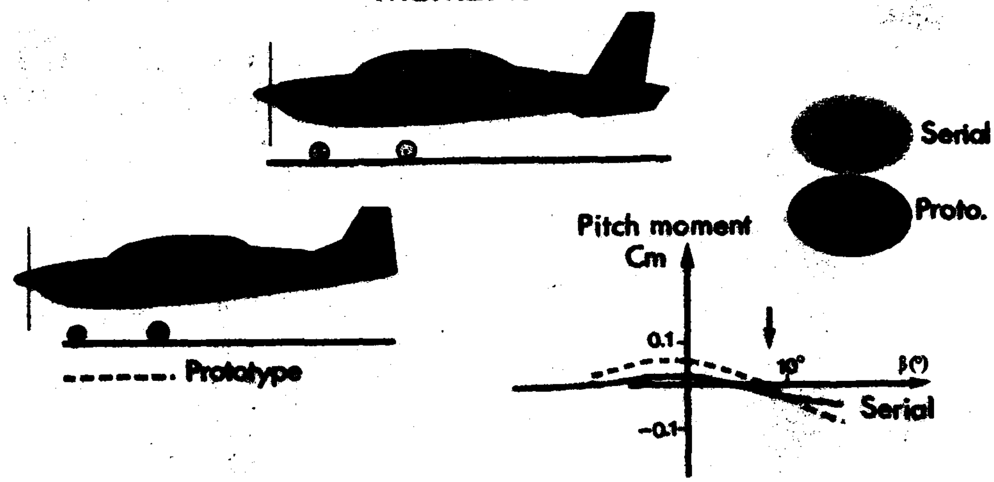


Figure: 15

AVION DE TRANSPORT REGIONAL

SIDESLIP EFFECT ON THE PITCHING MOMENT

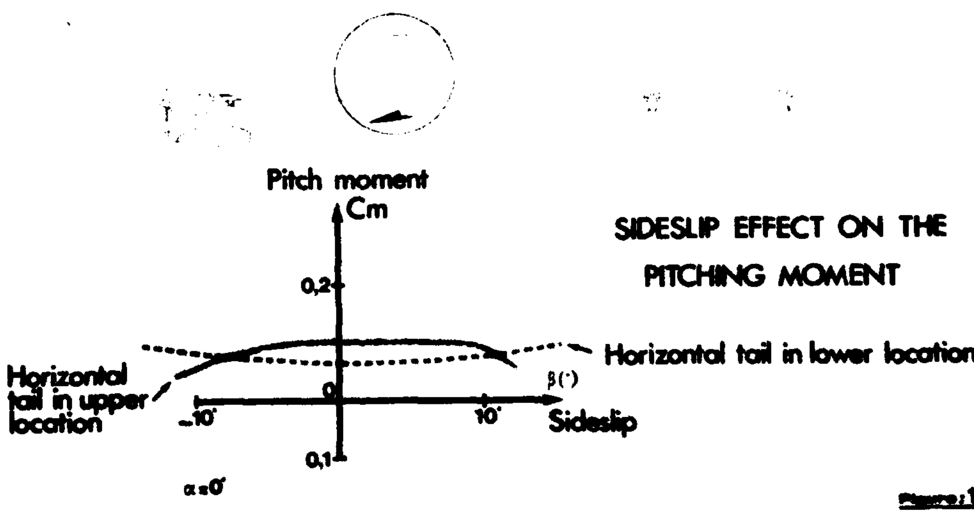


Figure: 16

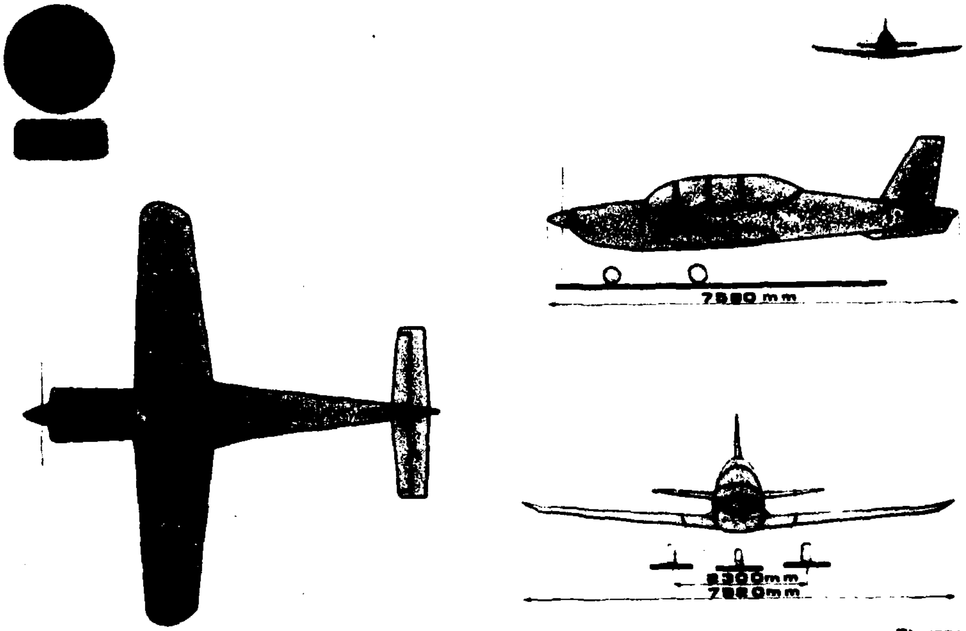


Figure: 17

TEST OF THE CRITERION

[illegible]

4 10^{-3}	4 10^{-3}	$\frac{\rho s \ell}{2m}$
3	4	$-C_{nr}$
0.7	0.9	$C_{n\beta}$

$$\frac{\rho s \ell}{2m}\left(\frac{\rho_y}{\ell}\right)^2\left[C_{z\alpha}+\left(\frac{\ell}{\rho_y}\right)^2 C_{mq}\right]\left[C_{y\beta}+\left(\frac{\ell}{\rho_r}\right)^2 C_{nr}\right] C_{n\beta}$$

$C_{n\alpha}$ x [illegible]

0.12	**0.2**	
0.25	**0.05**	
0.4	0.5	[illegible] $\frac{\ell_d}{\ell}$
1.25	1.6	[illegible] $\left(\frac{\ell_d}{\ell}\right)^2$

Figure: 18

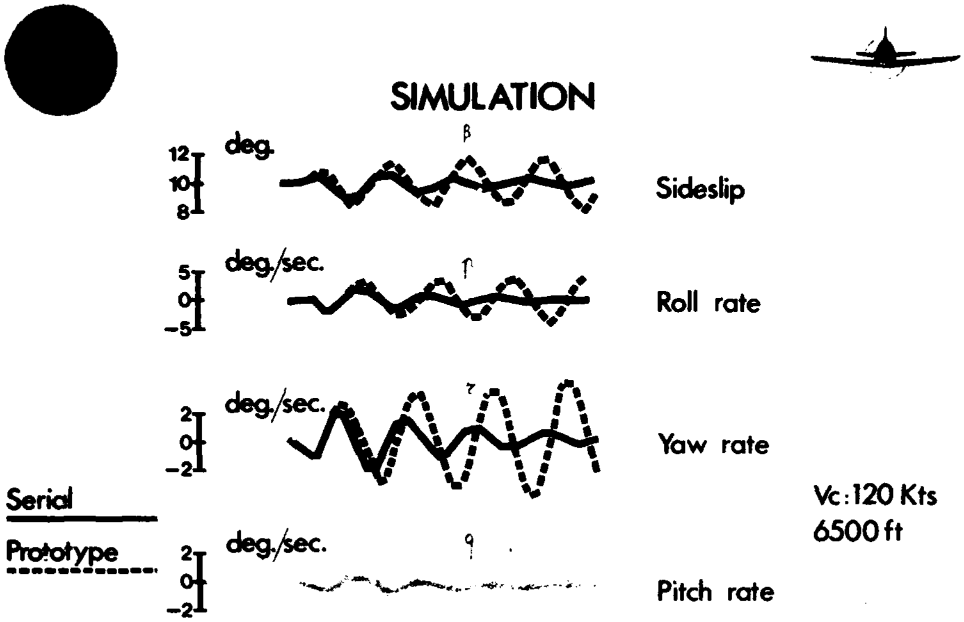


Figure: 19

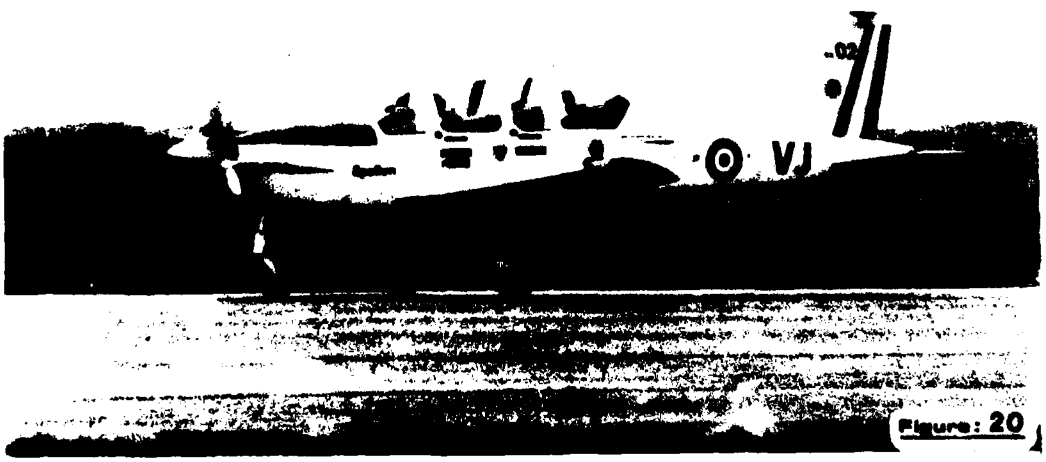


Figure: 20

ADVANCED FLIGHT CONTROL DESIGN TECHNIQUES AND HANDLING QUALITY REQUIREMENTS*

THOMAS B. CUNNINGHAM
RHALL E. POPE

HONEYWELL SYSTEMS AND RESEARCH CENTER
P.O. BOX 321
MINNEAPOLIS, MN 55440
UNITED STATES OF AMERICA

1.0 INTRODUCTION

The piloting tasks of today's multi-surface high performance aircraft are becoming increasingly complex. Accommodating multiple axes dynamic coupling has become more difficult with the addition of direct force control and promises to become more difficult still with the introduction of thrust vectored propulsion systems. The additional capabilities provided by direct force and thrust vectoring while increasing weapon system effectiveness have introduced new requirements for handling quality specifications. Central to the adequate treatment of these new requirements are specifications which address the key element of multi axis dynamic coupling and high level of augmentation in the flight control system. These two elements seriously challenge the utility of current handling quality criteria defined in Mil-F-8785B and C and in Mil-F-9490.

Handling qualities criteria as specified in Mil F-8785 B/C are defined primarily from the aircraft transient response perspective. This assumes single input types of stimuli and that eigenvalues of linear fixed point models (either closed or open loop) are good indicators of the quality of response. Gain and phase stability margins as defined in Mil F-9490 assume decoupled loops which can be analyzed accurately one loop at a time. Classical frequency response techniques developed by Bode [1] and others are the basis for this criteria.

The introduction of multi-input coupled dynamics poses a problem for both of these criteria. Multi-mode command augmentation systems with high levels of augmentation have introduced new modes of response which do not easily fit into 8785 criteria. The notion of equivalent systems has been conceived to help address this issue. Dynamic coupling in multiple control loops has also made the application of classical techniques very difficult. Stability margins as defined by 9490 can also be misleading for systems with significant dynamic coupling.

Over the last two decades, modern control techniques have offered the promise to relieve the design problems accompanying multi-input dynamically coupled systems. Optimal control synthesis techniques, primary the Linear Quadratic Gaussian (LQG) approach, are structured to directly address the multi-input transient reponse design problem.

Control designs resulting from LQG synthesis were also initially thought to have very attractive stability robustness properties[2]. Additional research determined that these properties held only for the "full state" measurement case [3] which is difficult to achieve in practice.

More recently, renewed interest in robustness has spurred a rediscovery of the frequency domain as an insightful medium for analyzing multi-input system performance and stability characteristics. Additionally, recent research has determined that bandwidth may be a good indicator for specifying handling qualities for 6DOF systems [4]. The use of a "bandwidth hypothesis" criteria greatly facilitates design of guidance and control augmentation systems because similar criteria can be used for either manual or automatic loop closing.

The remainder of this paper is devoted to how modern control analysis and synthesis techniques can be applied in the frequency domain and likewise tied to the bandwidth considerations for handling quality criteria and ride quality. Section 2 contains the control perspective for multi-input multi-output systems. Section 3 discusses the use of linear quadratic guassian (LQG) design techniques to meet frequency domain design goals. An example of a YF-4 (F-4 with horizontal cannards) is presented in Section 4. Conclusions and recommendations are made in Section 5.

2.0 CONTROL PERSPECTIVE

The fundamental goals of control of physical plants are:

- o Desired Command Response
- o Disturbance Rejection

* The authors wish to thank the staff of the Systems and Control Sciences Group at the Honeywell Systems & Research Center. Particular thanks goes to Dr. Joseph E. Wall and Mr. Stephen G. Pratt who assisted in the design example. Research supporting the concepts discussed herein was performed on numerous internal research projects, US Office of Naval Research contract N00014-75-C-0144 and US Department of Energy contract ET-78-C-01-3391.

We can add the basic plant stability to this list if the uncontrolled plant is unstable.

The problem is to achieve these specified performance objectives from an incompletely known aircraft in the face of uncertain disturbances. Such a problem invariably forces the use of feedback. The effect of feedback on aircraft performance is such an important element of this approach that a review of the fundamentals is required. By examining the properties of feedback in a general setting, without regard for the technique used to generate the feedback law, the fundamental relations will become evident. We will be highlighting the aircraft flight control problem, but utilize more general concepts described in mathematically rigorous detail in reference 5. We start with a discussion of Flying qualities and Ride quality.

2.1 Flying Qualities--MIL 8785C [6] provides the basic specifications for good flying qualities. For command reponse a popular interpretation of 8785 involves requiring the C* response [7] to fall within the envelope of Figure 1. This provides the command response objective which will be fulfilled by the control system. This is a well-accepted criterion for rigid aircraft.

The envelope of acceptable time-domain response shown in Figure 1 can be represented also in the frequency-domain. The control system must ensure that the actual aircraft response falls within this envelope. To accomplish this objective, we use both feedback and precompensation. In the frequency-domain, the effects of feedback and precompensation are readily understood and separated. As discussed in detail in sections 2.3 and 2.4, feedback can reduce the uncertainty associated with our nominal open-loop model over frequencies of interest. In other words, it "shrinks" the envelope of frequency responses. We must use enough feedback that the envelope of closed-loop responses is smaller than the design envelope. The size of these envelopes yield frequency-domain bounds on the "size" of the loop gains. Precompensation is then used to adjust this envelope so that it falls within the design envelope. In essence, precompensation performs a band-limited inversion of the closed-loop system and then inserts the desired dynamics. Precompensation can be successful only after sufficient feedback has been applied. The feedback design process is explained in Section 3.

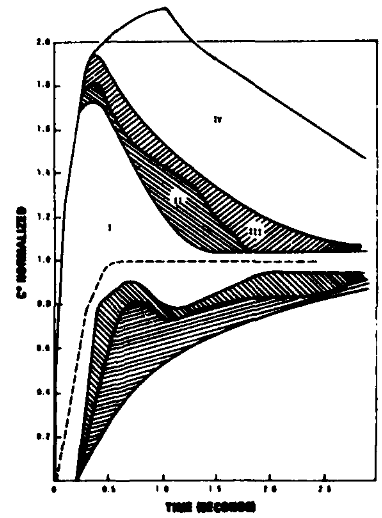


Figure 1. C* Envelope

2.2 Ride Qualities--There are a number of ride quality indices that are expressed as rms values, as limits on power spectral density responses (PSD) or as weighted integrals of PSD. The U.S. Air Force crew ride index is presented in MIL-F-9490 [8]. A discomfort index D_I is defined as:

$$D_i = \left[\int_{0.1}^{f_t} |W(f)|^2 \, |T_{CS}(f)|^2 \, \Phi_u(f)\,dt \right]^{1/2} \qquad (1)$$

where

D_i	=	Ride discomfort index (vertical or lateral)
$W(f)$	=	Acceleration weighing function (vertical or lateral) 1/g, Figure 2.
$T_{CS}(f)$	=	Transmissibility at crew station, g/ft/sec
$\Phi_u(f)$	=	Von Karman gust power spectral density of intensity (vertical or lateral gust) specified in MIL-F-8785
f	=	Frequency, Hz
f_t	=	Truncation frequency (frequency beyond which aeroelastic responses are no longer significant in turbulence)

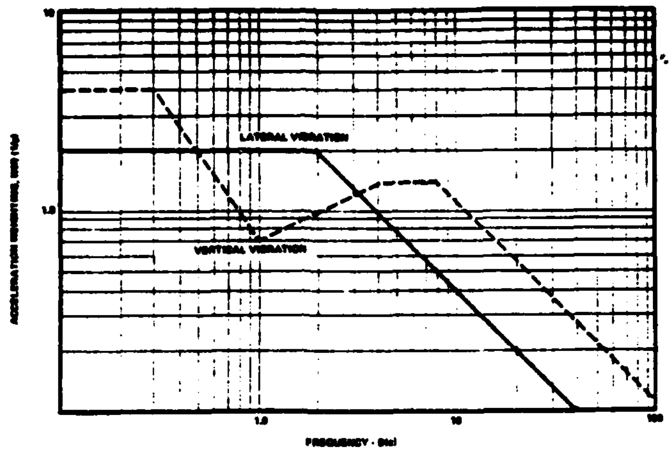


Figure 2. Acceleration Weighing Functions [8]

For design purposes, the discomfort index can be used to generate bounds on the loop gains to achieve acceptable disturbance rejection.

2.3 Feedback Improves Performance--A typical single-loop feedback situation for C* response is shown in Figure 3. Here we have disturbances acting on an aircraft to upset the goal of tracking the command C^*_C. We can express this goal as maintaining C* errors smaller than a specified level, ε,

$$|C^* - C^*_C| \leq \varepsilon \quad (2)$$

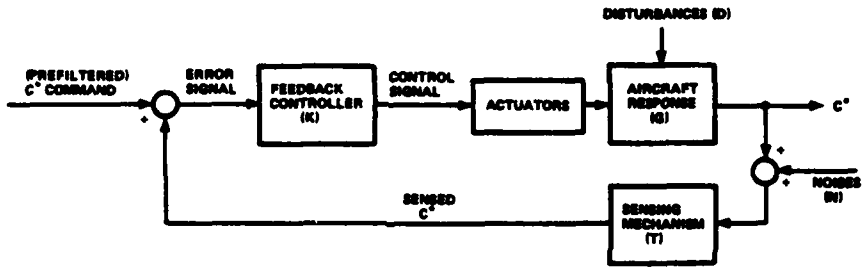


Figure 3. C* Feedback Loop

It is assumed that measurements of C* are imperfect because of sensor dynamics and noise. In typical feedback fashion, the sensed response is compared to the desired or commanded value, and the error is used by a controller to generate a control signal to drive the actuators on the aircraft.

Each of the elements of this loop can be represented by its transfer function relating the LaPlace transform of the response to the LaPlace transforms of the forcing function. The error response of the aircraft to commands, C^*_C, disturbances, D, and sensor noise, N, can then be developed by standard feedback equation manipulation.

Letting

$D_O = G_D D$ = Disturbance as seen at the output C* in the absence of feedback,

L = GKT = Feedback loop transfer function (i.e., transmission around the feedback loop) and

$\Delta T = T_{true} - T$ = C* sensor uncertainty we get:

$$(C^*-C^*_C) = \frac{1}{1+L}(D_O - C^*_C) - \frac{L}{1+L}\,\frac{(N+\Delta T C^*_C)}{T} \quad (3)$$

This equation relates system performance to each error source. Its various terms can be interpreted either as amplitudes of sine waves or, more generally as signal spectra. In either case, four immediate consequences can be seen:

Consequence No. 1. The loop transfer function must be large to achieve small errors. This follows from the first term on the right hand side of the error equation. In fact, in order to meet our specified error level, we must have:

$$|1+L| > \frac{|D_O - C^*_C|}{\varepsilon} \quad (4)$$

Thus, at those frequencies where either the disturbance responses or the commands are large compared with ε, we require L to be large.

Consequence No. 2. Model uncertainties (errors in L) must be small enough. This follows from both terms. Letting errors in L be ΔL, these terms require that $1 + L + \Delta L \neq 0$ for all frequencies and all ΔL. One way to ensure that this is true is to require $|1 + L| > |\Delta L|$ for all frequencies and all ΔL. This is a statement of the robustness of feedback systems and quantifies the amount of uncertainty which can be tolerated without loss of stability.

Consequence No. 3. Sensor noise must be small enough. This follows from the second term, which requires that:

$$\left|\frac{N}{T}\right| < \epsilon \quad \text{whenever } L \gg 1 \tag{5}$$

Consequence No. 4. Sensor uncertainties (errors in T) must be small enough. This also follows from the second term, which requires that

$$\left|\frac{\Delta T}{T}\right| < \frac{\epsilon}{|C^*_c|} \quad \text{whenever } L \gg 1 \tag{6}$$

This result specifies the uncertainty which can be tolerated in the C* measurement at each frequency.

For various applications, different ones of these consequences are are of dominant importance.

2.4 Multi-inputs and Multi-Outputs--In the minds of most engineers, these fundamental consequences of feedback are associated with the classical single-input single-output feedback theory. It is not generally recognized that they are equally true for more complex multi-input multi-ouput situations. This has been made clear by recent research on extensions of classical concepts to multivariable problems [5].

Consequence No. 1. $\underline{\sigma}(I+L) > \dfrac{||D_o - C^*_c||}{\epsilon}$

Consequence No. 2. $\underline{\sigma}(I+L) > \overline{\sigma}(\Delta L)$

Consequence No. 3 $||T^{-1} N|| < \epsilon$

Consequence No. 4. $\overline{\sigma}(T^{-1} \Delta T) < \dfrac{\epsilon}{||C^*_c||}$

here $\overline{\sigma}(\cdot)$ and $\underline{\sigma}(\cdot)$ are the maximum and minimum singular values of the indicated matrices and $||\cdot||$ represents the magnitude (norm) of the indicated vector. $\sigma(\cdot)$ can be interpreted as the maximum gain which the matrix can produce (at a frequency) and $\underline{\sigma}(\cdot)$ can similarly be interpreted as the minimum gain*. These details are discussed in more detail in reference 5.

2.5 Multiple Design Goals

In sections 2.3 and 2.4 we have discussed how to approach single and multiple input feedback control for a C* design effort. To incorporate ride quality into the design we must expand the feedback structure shown in Figure 3 to a vector performance goal. This is shown in Figure 4.

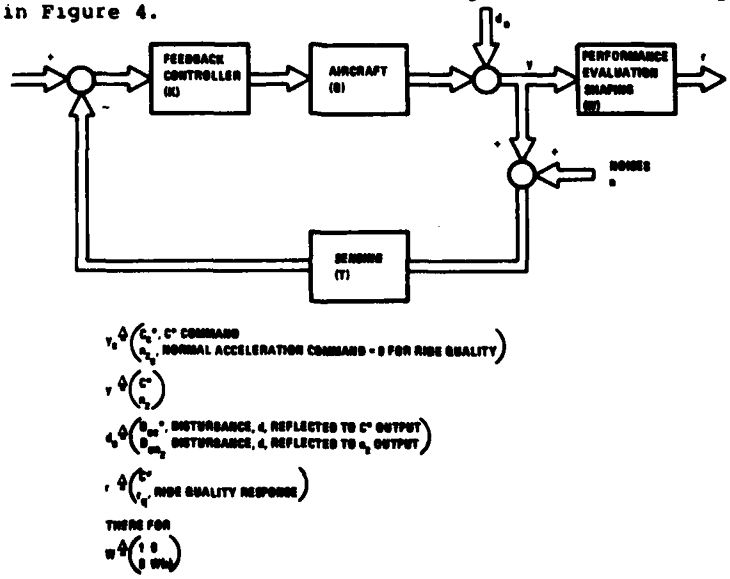


Figure 4. Multiobjective Design Loop

*Mathematically, we can express this as $\overline{\sigma}(A) = \max_{||x||=1} \dfrac{||Ax||}{||x||}$ and $\underline{\sigma}(A) = \min_{||x||=1} \dfrac{||Ax||}{||x||}$

The performance response weighting and n_z is $w(s)$. This is equivalent to the "acceleration weighting function" for vertical vibration of Figure 2.

Working these into our analysis we arrive at an expansion on our "consequences of feedback."

Consequence No. 1

$$\underline{\sigma}(I+L) > \max_{\omega} \; \frac{||D_{oc}^{*}||}{\epsilon_{c^*}} , \; \frac{||C_c^{*}||}{\epsilon_{c^*}} , \; \frac{||w(s)\, D_{onz}||}{\epsilon_{rq}}$$

Consequence No. 2

$$\underline{\sigma}(I+L) > \overline{\sigma}(\Delta L)$$

Consequence No. 3

$$||\, T_n^{-I} || < \min_{\omega} \{\epsilon_{c*}, \epsilon_{rq}/||w(s)||/ \quad \text{whenever } L >> 1$$

Consequence No. 4

$$\overline{\sigma}(T^{-1}\Delta T) < \frac{\epsilon_{c*}}{||C_c^{*}||}$$

2.6 Graphical Interpretation--There is no need to delve into the intricacies of matrix theory in order to understand the fundamentals of feedback. Performance can be shown graphically by plotting the right hand side of consequence no. 1.

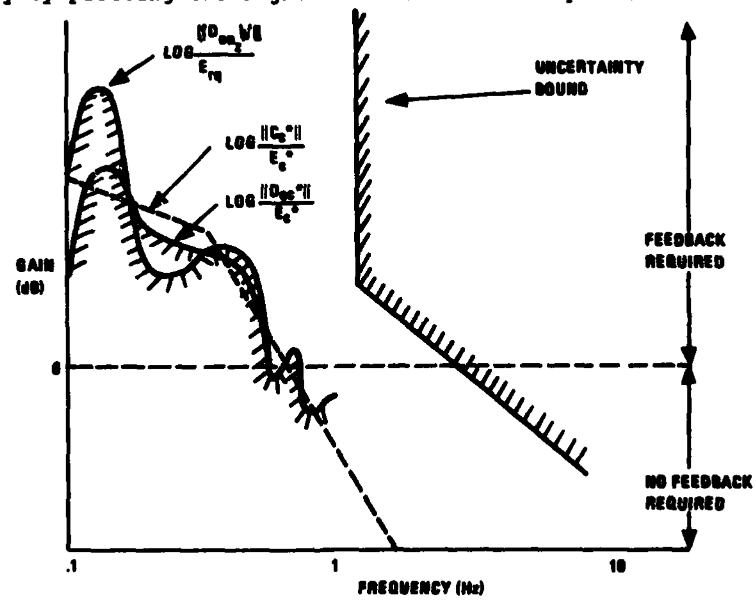


Figure 5. Design Objectives

Such a plot is shown in conceptual form in Figure 5. Magnitude vs. frequency on a log-log scale has been plotted. Wind gust disturbances have been plotted according to their effect on the C* output. Command spectra are included as the transform of the desired response to C*-commands. Likewise, a ride quality boundary plot completes the right hand side of consequence no. 1.

If the 0dB line is inserted on Figure 5, Consequence No. 1 can be used to determine requirements on L. In particular, for those parts of the figure which lie below this line, no feedback is required, i.e. $\overline{\sigma}(L)$ can be less than 1 at those frequencies. However, for those parts which lie above the line, we require $\underline{\sigma}(L) > 1$ in order to reduce errors to the ϵ-level in the closed loop.

Consequence No. 2 also can be interpreted graphically. We recognise that for any physical system, the uncertainty associated with a model of that system inevitably becomes large at high frequencies. In order to ensure stability, the feedback loop must roll-off before the magnitude of the uncertainty reaches unity and it must attenuate quickly enough. This attenuation rate is the uncertainty associated with our design model and is plotted in Figure 5. Stability requires that $\overline{\sigma}(L)$ lies below this curve.

These observations suggest that Figure 5 can be used to determine graphically the suitability on any L, and thus of any compensator K, by directly sketching the singular values of L. Such a plot is shown as Figure 6. We observe that it is desirable to

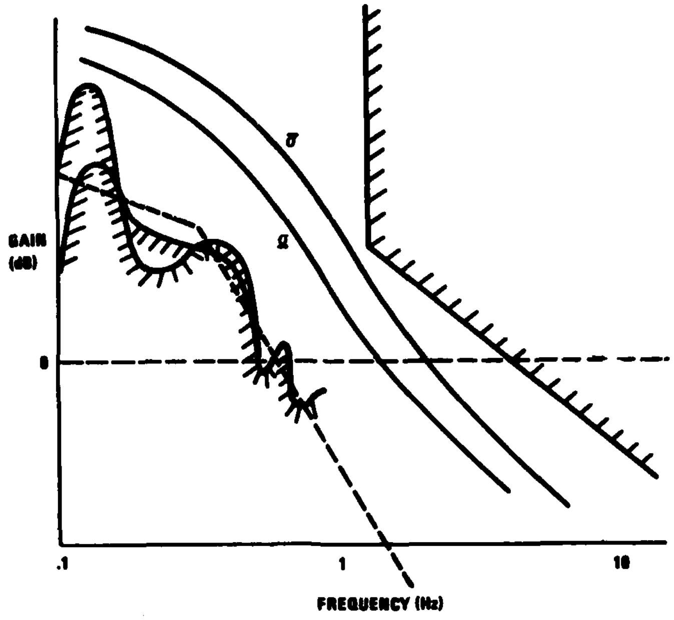


Figure 6. Trial Design

have the σ(L) lines follow as closely as possible to the disturbance or command limits. This minimizes control authority. Also, we have a prior assessment of component bandwidths in terms of performance specifications and the disturbance/command environment.

This discussion has addressed fundamental feedback relations which must be obeyed by any control system. These relations hold for all controllers whether designed with modern or classical tools. These fundamentals relate the size of disturbances and commands to the size of the required control law. They provide a multivariable generalization of classical single-loop concepts such as large loop gains give good performance. To be successful, a design technique must manipulate the shape of the loop gains. The use of LQG synthesis to provide these desired loop shapes is presented in the next section.

3.0 CONTROL SYSTEM DESIGN

3.1 Feedback Design Objective--The design objective of a feedback control system for advanced aircraft can be expressed in the frequency-domain in terms of properties of the loop transfer matrix. In Section 2, we explained how model uncertainties and reduced-order models impose limitations on the loop gains. Specifications on handling qualities and ride qualities were shown to translate into requirements on the loop gains. Summarizing these discussions, the feedback design problem becomes one of finding a compensator K(s) which shapes the loop transfer matrix G(s)K(s) in such a way that:

1. The loop gains $\underline{\sigma}(GK)$ are high at low frequencies to meet the handling qualities and ride qualities performance requirements.
2. The loop gains $\bar{\sigma}(GK)$ are low at high frequencies to meet stability robustness requirements, and
3. The transition, or "crossover," between these two regions is accomplished in a stable manner.

An important discussion of this third point is found in reference [5]. This frequency domain interpretation of the design problem was illustrated graphically by the multivariable σ-plot shown in Figure 5.

So far, we have described the feedback design problem as a design tradeoff involving performance objectives and stability requirements. This tradeoff is essentially the same for SISO and MIMO problems. Design methods to carry it out, of course, are not. For scalar design problems, well-developed tools (i.e., "classical control") exist which permit designers to construct good transfer functions for Figure 5 with relatively little difficulty. Various attempts have been made to extend these methods to multivariable design problems: "Single-loop-at-a-time" methods, the Inverse Nyquist Array [9] methodology, and the Characteristic Loci [10] methodology. These methods are based on the idea of reducing the multivariable design problem to a sequence of scalar problems. It is shown in Reference [5] that these design approaches are not generally reliable in achieving the design objectives shown in Figure 5. The difficulty is that the selected set of scalar design functions are not necessarily related to the system's actual feedback properties. Further elaboration of this point and an example illustrating the problem may be found in [5].

A second major approach to multivariable feedback design is the modern Linear-Quadratic-Gaussian (LQG) procedure. Honeywell has been deeply involved in research in this area and has successfully applied the methodology to numerous aerospace systems, from the 1968 flight tested B-52 LAMS Program to the C-5A ALDCS, the YF-12 LAMS, the J-85 engine, and the F-4 lateral axis, the F-8C CCV, the HIMAT vehicle and Re-entry Guidance and control. These experiences have brought a fundamental reinterpretation of the LQG methodology. No longer is the methodology viewed as a way to optimize time responses. Rather, it can be as an effective tool to do multivariable control design in the frequency-domain. This new LQG/frequency domain philosophy is discussed in the following.

3.2 LQG Loop-Shaping

A detailed description of the manner in which LQG can be used to solve multivariable frequency-domain problems is given in the reference [5]. The properties of LQG loops which make effective for frequency-domain design are briefly summarized below.

We will deal with the standard LQG controller configuration shown in Figure 7. This has the same structure as the generic control system introduced earlier in Figure 4. In this figure, the controller is treated as an ordinary finite dimensional linear compensator with a special internal structure consisting of a Kalman-Bucy filter (KBF) cascaded with a linear-quadratic state feedback regulator (LQR). Standard symbols will be used to represent these elements; i.e., the state space realization of G(s) is composed of the matrix triple A, B, C, the control gains are denoted by matrix K_c, the filter gains by K_f, and the weighing and noise intensity matrices by Q, R and Ξ, θ, respectively.

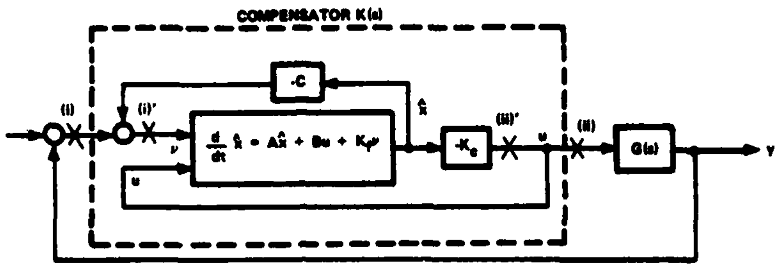


Figure 7. The LQG Feedback Loop

The loop transfer properties of interest are those corresponding to loop-breaking points (i) and (ii) of the figure. Two other loop-breaking points, (i)' and (ii)' also are shown in the figure. These are internal to the compensator and therefore have little direct significance. However, they have desirable loop transfer properties which can be related to the properties of points (i) and (ii). We will concentrate on points (ii) and (ii)' and present the theory for these points. We return to points (i) and (i)' at the end of this section.

FACT 1. The loop transfer function obtained by breaking the LQG loop at point (ii)' is the LQR loop transfer function $K_c(sI-A)^{-1}B$.

Fact 2. The loop transfer function obtained by breaking the LQG loop at point (ii) is KG. It can be made to approach the LQR loop transfer function $K_c(sI-A)^{-1}B$ arbitrarily close by designing the KBF according to a full-state loop transfer recovery procedure due to Doyle and Stein [11].*

The significance of these facts is that we can design LQG transfer functions on a full-state feedback basis and then approximate them adequately with a recovery procedure. This robustness procedure has been demonstrated for wing flutter control [12], and RPV control design [13].

Full-State Loop Transfer Design--The intermediate full-state design step is worthwhile because LQR loops have good classical properties [14]-[16]. The basic result is that LQR loop tanfer matrices $T(s) = K_c(sI-A)^{-1}B$ statisfy the following return difference identity [12] for all frequencies ω:

$$[I+T(j\omega)]^*R[I+T(j\omega)]=R+[H(j\omega I-A)^{-1}B]^*\ [H(j\omega-A)^{-1}B] \qquad (7)$$

where $H^TH=Q>0$ is the state weighting matrix. When the control weighting matrix is a scalar times the identity, $R=\rho I$, the singular values of the transfer function T satisfy**

---

* This requires the assumptions that G(s) is minimum phase and has at least as many outputs as inputs

** Non-identity R matrices can be substituted in B by letting B' = BR/2.

$$\sigma_i[T(j\omega)] \simeq \sigma_i[H(j\omega I-A)^{-1}B]/\sqrt{\rho}$$

whenever the right-hand-side is much greater than 1[5]. This means we can choose ρ and H explicitly to satisfy the low frequency performance conditions shown in Figure 5.*

The return difference identity (6) also guarantees that the LQR return difference always exceeds 1, i.e., $\underline{\sigma}[I+T(j\omega)]\geq 1$ for all frequencies. This is a multivariable generalization of avoiding the -1 critical point on the Nyquist Diagram for scalar systems. It implies that LQR loops provide resonable transition or "crossover" between the low and high frequency regions shown in Figure 5.

Finally, we note that at high frequencies the LQR loop approaches

$$\bar{\sigma}[T(j\omega)] \simeq \frac{\bar{\sigma}[HB]}{\rho\omega}$$

This shows how the high frequency roll-off characteristics are related to ρ and H. This is a relatively slow attenuation rate and is the price the regulator pays for its excellent return different properties. We recognize that no physical system can maintain indefinitely a $1/\omega$ characteristic. This is not a concern since T(s) is only a design function and will be approximated by the full-state loop transfer recovery procedure.

<u>Full-State Loop Transfer Recovery</u>--As mentioned earlier, the full-state loop transfer function designed above for point (ii)' can be recovered at point (ii) by a modified KBF design procedure.** Special noise statistics are used for the KBF design. The drive noise intensity matrix is modified as $\Xi = q\, BB^T$ where Ξ is the nominal noise intensity matrix and q is a scalar parameter. Then as q becomes large, we know that the filter gain K_f behaves in such a way to yield loop transfer recovery for point (ii). Mathematically [13], $K(s)G(s) \rightarrow K_c(sI-A)^{-1}B$ as $q \rightarrow \infty$.

Based on the above summary of frequency-domain properties of LQG controllers, the following simple loop-shaping procedure is suggested.

Step 1. Design a LQG with ρ and H selected such that the loop transfer function T(s) meets performance and stability robustness requirements (Figure 5).

Step 2. Design a sequence of KBF's with modified driving noise intensity matrix and the parameter q allowed to take on consecutively large values.

Step 3. Select an element of the resulting sequence of transfer functions. $K(s)G(s) \rightarrow T(s)$, $q \rightarrow \infty$ which adquately approximates the desired functions over the frequency range of interest.

All design objectives including nominal stability are then assured. Our conclusion is that LQG represents a powerful, general tool for obtaining multivariable frequency-domain designs.

We have discussed designing a full-state regulator and then recovering the full-state properties with a Kalman-Bucy Filter. We followed this approach because this is the usual sequence one considers for LQG design. We note that the procedure also applies to loop-breaking point (i) in Figure 7. For this point, however, the role of the filter and the controller are reversed. We begin by designing a KBF whose loop transfer function $C(sI-A)^{-1}K_f$ (at point (i)') has good frequency domain properties. Then we design a sequence of LQR's which serve to recover this function at point (i) [5]. The equations for this alternate procedure are mathematical "duals" of the ones given above. The subtle differences between the two procedures are discussed in Ref. [17].

The dual procedure, designing the filter first and then recovering the full-state properties with regulator, is more compatible with the objectives of this paper and will be used for the control design task. It is illustrated in the following aircraft design example.

---

* It also may be necessary to append additional dynamics. For example, to achieve zero steady-state errors may require additional integrators in the plant. This is equivalent to "frequency-dependent" weighting.

** The required theoretical assumptions are that the plant have at least as many outputs as inputs and be minimum phase. In practice, the recovery procedure is effective as long as G(s) has no right-half-plane zeros below the crossover frequency. The limitations on the achievable performance of feedback systems because of non-minimum zeros is discussed in reference [15].

4. Aircraft Design Example

To illustrate the specification interpretations of section 2 and the LQG loop shaping methodology of section 3 a pitch axis control law design is presented in this section for a special version of the F-4 aircraft. Using data from reference 18 a single flight condition for the YF-4 aircraft (M=.6 at Sea level) is chosen. The YF-4 has twin horizontal canards which can be used for feedback control. The short period model with actuator dynamics in state space form is

$$\begin{bmatrix} \dot{\alpha} \\ \dot{q} \\ \dot{\delta}_s \\ \dot{\delta}_c \end{bmatrix} = \begin{bmatrix} -1.0504 & .9915 & .1232 & -.0362 \\ 10.80 & -1.328 & -16.434 & 6.34 \\ 0 & 0 & -23.0 & 0 \\ 0 & 0 & 0 & -20.0 \end{bmatrix} \begin{bmatrix} \alpha \\ a \\ \delta_s \\ \delta_c \end{bmatrix} + \begin{bmatrix} 0 & 0 \\ 0 & 0 \\ 23.0 & 0 \\ 0 & 20.0 \end{bmatrix} \begin{bmatrix} \delta_{sc} \\ \delta_{cc} \end{bmatrix}$$

where the state vector is

$$\begin{bmatrix} \alpha = \text{angle of attack - rad.} \\ q = \text{pitch rate - rad/sec.} \\ \delta_s = \text{stabilator actuator - rad.} \\ \delta_c = \text{canard actuator - rad.} \end{bmatrix}$$

and the input vector is

$$\begin{bmatrix} \delta_s = \text{stabilator command - rad.} \\ \delta_c = \text{canard command - rad.} \end{bmatrix}$$

The open loop vehicle is statically unstable with short period poles at -4.464, and +2.086. Outputs available through sensor measurement are pitch rate, q, and normal acceleration, at the pilot station, n_{zp}. Assuming no sensor dynamics or errors (noise will be addressed by proper attenuation) these outputs are related to the command specifications of C* and n_{zp} (for ride quality) as follows:

$$\begin{bmatrix} C^* \\ n_{zp} \end{bmatrix} = \begin{bmatrix} 324 & 1 \\ 0 & 1 \end{bmatrix} \begin{bmatrix} q \\ n_{zp} \end{bmatrix}$$

This transformation of the sensed outputs to command outputs properly poses the problem in the appropriate design form which is compatible with the block diagram in figure 4. This allows us to view the design goals highlighted in figures 5 & 6. For this example we have chosen to design a level 1 C* response (in addition to stabilizing the aircraft). Verification of ride quality is beyond the scope of this example.

The form of the control design problem requires desired loop shaping at the outputs of the plant. We must therefore use the dual design procedure of the steps outlined in section 3 and start with the Kalman-Bucy filter (KBF) design. Recall that the compensator, K(s), relates to figure 7 by $K(s) = K_c(sI-A+K_fC+BK_c)K_f$. The KBF design will design a portion of the compensator, $C(sI-A)K_f$, K_f, which is robust at loop break point (i)'. We design a loop which is robust at point (i) with the recovery step (the dual of step #4 in section 3) giving the control law gain, K_c.

For this simple example we have chosen the two singular values to have identical 1/s roll off from .1 r/s to 100 r/s frequencies and crossover at 4 r/s. Applying this design and using the Honeywell experimental computer aided design package (HONEY-X) these goals are approximately, but readily, achieved for the KBF loop as chosen in figure 8. The difference between figure 8 and desired 1/s rolloff is caused by the requirement to stabilize the unstable vehicle with better than 6db negative gain margin [2].

Having designed K_f to shape the loops achieved in figure 8 we now must design the optimal quadratic control gain, K_c. As outlined in section 3. this is the dual of the Doyle robust estimator [11]. This involves sequentially lower quadratic weighting of the control inputs in the LQR performance index.

Without getting into the intricacies of state space modeling and Ricatti equations this exercise can be summarized in figure 9. Figures 9a through 9c show the recovery process for successively lower control weights.

Figure 10 shows the final feedback loop shape. This result was achieved after compensator order reduction and additional low pass filtering applied to the 9c design. Specifically, the LQG design produced a sixth order compensator, K(s), i.e., four states for the aircraft and two for integral control at low frequency in both loops. Three of the compensator roots were pushed to high frequency in the design

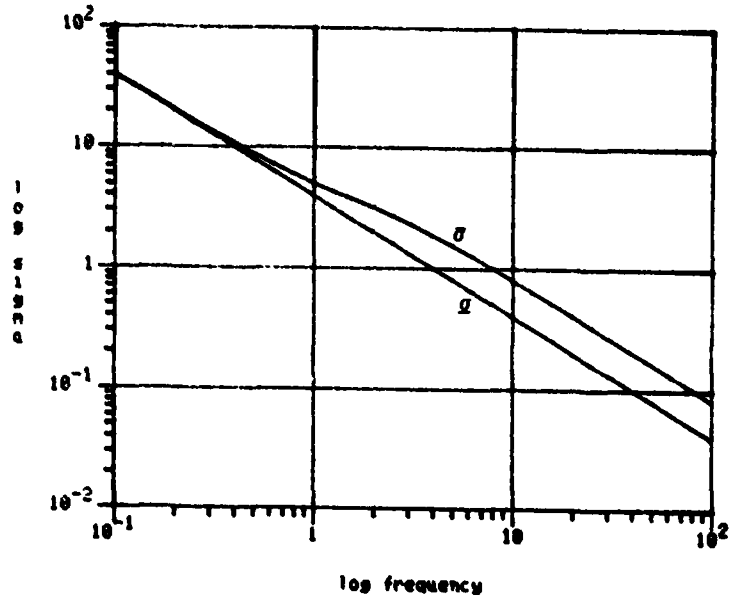
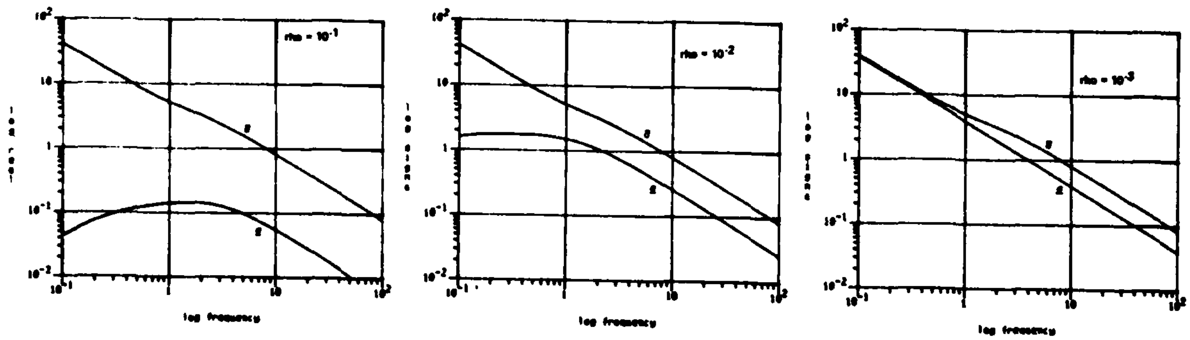


Figure 8. State KBF Design



9a. Low Gain K_c 9b. Moderate Gain, K_c 9c. High Gain K_c

Figure 9. Recovery Procedure

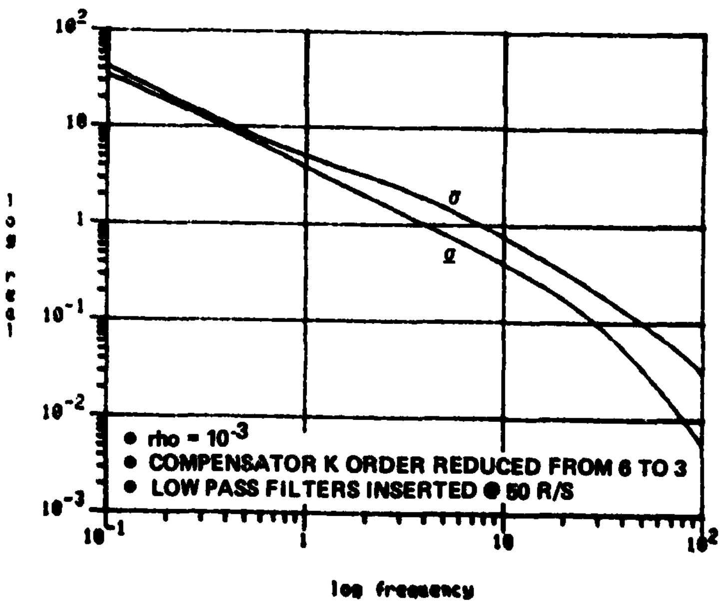


Figure 10. Final Control Law, GK

process. These can be eliminated through residualization. Finally, additional rolloff to eliminate sensor noise and achieve added robustness to high frequency bending modes was achieved by adding a 50 rad/sec low pass filter to each sensor loop. The point here is that - <u>when properly designed in the frequency domain LQG does not produce complicated, high order, compensators.</u>

Consequence #2 in section 2 contains a robustness requirement. Constructing the appropriate uncertainty model ΔL is difficult and was not attempted here. The singular values of I+GK do provide robustness insight, however. Figure 11 shows the I+GK singular values for this design case. Notice that at $\sim\omega$=11 rad/sec. we have $\underline{\sigma}$=.7. This can be viewed as the minimum distance to a multi-input multi-output critical point (-I) and has the same useful interpretation as gain and phase margins have in the Nyquist plane for single loop systems. For example, $\underline{\sigma}$=.7 might be interpreted as a gain margin of 10.5db (if we conservatively assume a phase of -180°) or a phase margin of 41° (if we conservatively assume a gain of 1).

Such interpreations are mathematically meaningless. However, the point is that gain and phase margins are derived from the notion of distance from the critical point. Proponents of Nyquist plane and Nichols charts interpretations of stability margins implicitely know this. The minimum closed loop singular value, $\underline{\sigma}$, of I+GK is merely a multiloop extension of this basic idea.

Finally, knowning that we are robust in frequency to approximately 2 hertz the final C* design is completed with the appropriate precompensation design for the pilot stick input. Here the best approach is to examine the C_c^* to C* closed loop transfer function. For this design the transfer function is.

$$\frac{C^*}{C_c^*}(s) = \frac{(S/.02+1)(S/1.06+1)(S/3.89+1)(S/5.28+1)(S^2/89.1+S/17.1+1)}{(S/1.7+1)(S/4.17+1)(S/4.77+1)(S^2/109.4+S/5.9+1)(S/20.35+1)(S/41.8+1)}$$

A pilot stick shaping feedforward transfer function of

$$\frac{C^*_c}{\delta_s}(s) = \frac{(.75s+1)}{(s+1)(s/10+1)}$$

produces a level 1 C* response shown in figure 12 plus good pilot input attenuation.

4.0 CONCLUSIONS

Conceptual descriptions of U.S. military specifications, C* and ride quality, have been given in the frequency domain. The control law designer can use his set of linear analysis tools to analyze and design control laws to provide the appropriate balance between performance requirements and tolerance to existing model uncertainty. New extensions of classical frequency domain tools for single input single output analysis to multi-input multi-output systems have been discussed based on singular values. Singular value loop shaping via a new interpretation of "Linear-Quadratic-Guassian" optimization is discussed. Finally, an example of the analysis and design principles is presented with the assistance of the HONEY-X software package.

Based upon these findings numerous conclusions can be drawn:

- o Frequency domain interpretation of flying quality specifications are very meaningful to the control law designer. Guides for such interpretations (as conceptualized here) should be developed.
- o Stability margins for multi-input multi-output systems based upon matrix size measures, such as singular values, should be developed.
- o Multi input synthesis tools need further development. LQG has demonstrated its usefullness, however, more loop shaping properties need to be developed.
- o We are past the age of analog computers and pencil and paper designs of single loop systems. Sound, numerically stable, analysis and synthesis algorithms are required to utilize the control law design methods described herein. CAD tools such as HONEY-X are emerging - these should be expanded.
- o Future updates to military specification to incorporate ideas such as described here should be examined jointly by the handling quality and control design specialists.

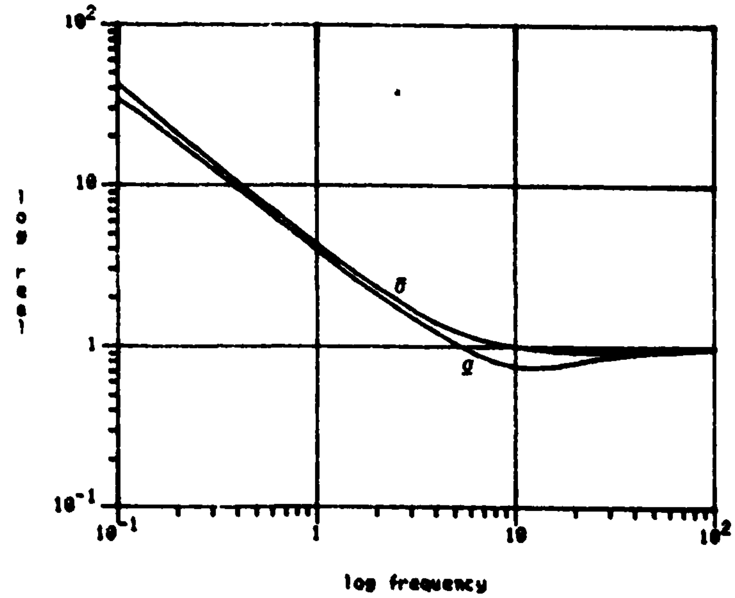


Figure 11. Singular Values of (I+GK)

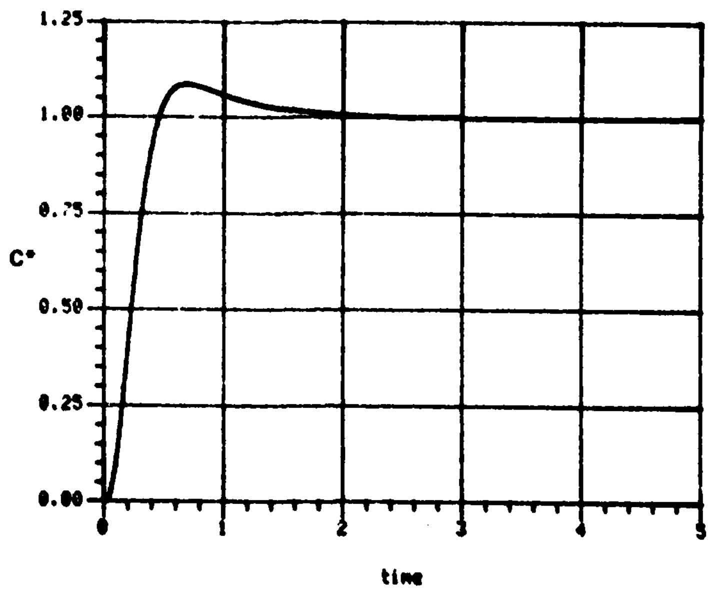


Figure 12. Closed Loop C^* Response

REFERENCES

1. Bode, H.W. Network Analysis and Feedback Amplifier Design, D. Van Nostrand, Princeton, 1945.
2. Kalman, R.E., "When is a Linear System Optimal?," Transactions ASME Series D: Journal of Basic Engineering, Vol. 86, pp. 51-60, 1964.
3. Doyle, J.C., "Guaranteed Margins for LQG Regulators," IEEE Transactions on Automatic Control, Vol. 23, No. 4, August, 1978.
4. Hoh, R.H., et. al., "Development of Handling Quality Criteria for Aircraft with Independent Control of Six Degrees of Motion," AFFDL-TR-81-3027, April 1981.
5. Doyle, J.C., and Stein, G., "Multivariable Feedback Design: Concepts for a Classical/Modern Synthesis," IEEE Transactions on Automatic Control, Vol. 26, No. 1, pp. 4-16, February, 1981.
6. Anon., "Military Specification Flying Qualities of Piloted Aircraft," MIL-F-8785C, Nov. 1980.
7. Anon., "New Short-Period Handling Quality for Fighter Aircraft," Boeing document D6-17841T/N, Oct. 1965.
8. Townsend, J.L., and Raymond, E.T., "Background Information and User's Guide for MIL-F-9490," AFFDL-TR-74-116, Jan. 1975.
9. Rosenbrock, H.H., Computer-Aided Control System Design, Academic Press, 1974.
10. MacFarlane, A.G.J., and Kouvaritakis, B., "A Design Technique for Linear Multivariable Feedback Systems," Int. Journal of Control, Vol. 25, pp. 837-879, 1977.
11. Doyle, J.G., and Stein, G., "Robustness with Observers," IEEE Transactions on Automatic Control, Vol. 24, No. 4, Aug. 1979.
12. Mahesh, J.K., Stone, C.R., Garrard, W.L., and Hausman, P.D., "Active Flutter Control for Flexible Vehicles," NASA CR 159160, Nov. 1979.
13. Hartmann, G.L., Stein, G., and Greene, C.S., "LQG Controls for Highly Maneuverable Aircraft," AIAA Atmospheric Flight Mechanics Conference, August, 1981.
14. Kalman, R.E., "When is a Linear System Optimal?," Trans. ASME Ser. D: J. Basic Eng., Vol. 86, pp. 51-60 (1964).
15. Anderson, B.D.O. and Moore, J.B., Linear Optimal Control. Prentice-Hall, 1971.
16. Safonov, M.G. and Athans, M., "Gain and Phase Margin of Multiloop LQG Regulators," IEEE Trans. Auto Control, April 1977.
17. Wall, J.E., Harvey, C.A., and Doyle, J.C., "Tradeoffs in the Design of Multivariable Feedback Systems," Eighteenth Allerton Conf., 10/8-10/80.
18. Anon., "Advanced Fighter Digital Flight Control System (DFCS) Definitin Study" Honeywell Report, P.O. #Z40020, March 1975.

ANALYSE DU ROLE DES ASSERVISSEMENTS POUR UN AVION SUBSONIQUE A STABILITE LONGITUDINALE REDUITE

F. IANNARELLI

AEROSPATIALE
316, route de Bayonne
31060 TOULOUSE - FRANCE

SOMMAIRE

Introduits essentiellement pour restituer des qualités de pilotage acceptables à des centrages arrière, les asservissements de la commande de profondeur, grâce à leurs nombreuses possibilités seront utilisés afin d'améliorer au maximum le confort du pilotage de l'avion.

Après avoir présenté successivement les différents types d'asservissements :
- élémentaires : dont le rôle est de moduler la position des points fondamentaux de l'avion,
- élaborés en pente : dont la fonction est d'assurer l'autotrim de l'avion ou le maintien de la pente,

une adaptation à un avion à stabilité longitudinale réduite du type AIRBUS sera fournie.

1. INTRODUCTION

La conception d'un avion de transport subsonique à stabilité longitudinale réduite permet d'escompter des gains en performances non négligeables, mais au détriment des qualités du pilotage d'un avion qui resterait équipé de commandes de vol classiques.

La recherche de la consommation minimale en croisière conduit à faire voler les avions à des centrages incompatibles avec des commandes de vol classiques. La restitution de qualités de pilotage acceptables nécessite l'introduction d'asservissements dans la commande de profondeur. Toutefois, si cette dernière peut être mécanique, les possibilités offertes par ces asservissements ne seront réellement exploitées qu'au travers de commandes de vol électriques.

2. RAPPEL DES PHENOMENES LIMITES RENCONTRES EN VOL

On peut déterminer pour un avion différents points caractéristiques, tels que :
- foyer en vitesse (Fv)
- point de manoeuvre (Fq).

Pour un avion de type AIRBUS (bimoteur sous voilure), dont la position de ces différents points caractéristiques en croisière est donnée sur la planche 1, le recul du centrage au-delà de ces points induira les phénomènes suivants :

- centrage de l'avion (XG) en arrière de Fv :
  - . avion instable en vitesse,
  - . couplage entre oscillation d'incidence et phugoïde avec apparition d'un mode apériodique pouvant être instable et dont le taux de divergence peut gêner le pilotage ;
- centrage proche du point de manoeuvre :
  - . diminution des braquages par g,
  - . augmentation des temps de réponse avion,
  - . augmentation du taux de divergence, soit une détérioration notable du pilotage.

Le positionnement du centrage en avant de Fv, imposé par des exigences réglementaires (stabilité de l'avion se traduisant par un effort au manche répondant au critère de la livre par 6 kts), peut ne pas être respecté dans une situation dégradée de pilotage après panne. Si on peut tolérer un certain niveau d'instabilité pour un mouvement à longue période, il n'en est pas de même pour un mouvement à période plus courte. Dans la pratique, cette situation qui ne se rencontre qu'aux centrages arrière, où après dégénérescence de la phugoïde et de l'oscillation d'incidence apparaissent des modes apériodiques instables, limitera la valeur maximale arrière du centre de gravité. Au-delà de cette limite le maintien en vol de l'avion, même pendant quelques minutes seulement, risque dans certaines conditions de ne plus être possible.

Nous verrons dans la suite de cette communication que l'on peut considérer également un autre point fondamental : le foyer en incidence ($F\alpha$), celui-ci permettant de caractériser l'amortissement de l'oscillation d'incidence.

L'on ne devra pas perdre de vue, lors de la conception de systèmes déplaçant à volonté les points caractéristiques (Fv, $F\alpha$, Fq), que compte tenu de l'éventualité de pertes partielles et totales, momentanées ou définitives de ces aides, le pilotage de l'avion dit "naturel" doit être pris en considération.

3. DEFINITION DES ASSERVISSEMENTS

Nous distinguerons deux types d'asservissements :
- élémentaires,
- élaborés.

3.1 Asservissements élémentaires

On appelle asservissements élémentaires les asservissements de la commande de profondeur :
- à la vitesse de tangage (q),
- à l'incidence (α)
- à la vitesse longitudinale de l'avion (V)
- à toute combinaison de ceux-ci (par exemple le facteur de charge (n_z)) utilisés tels quels, sans filtre, par simple braquage proportionnel, soit par exemple :

$$\delta q \text{ (braquage de la gouverne de profondeur)} = Kq.(q) + K\alpha.(\Delta\alpha) + Kv.(\Delta V)$$

Ces asservissements ont la propriété de conserver aux équations qui régissent le mouvement longitudinal de l'avion asservi, la même structure que celle des équations représentant le mouvement longitudinal de l'avion naturel.

Une formulation matricielle de cette structure est présentée ci-après :

$$\begin{bmatrix} \Delta\dot{V}/V \\ \dot{\gamma} \\ \dot{\alpha} \\ \dot{q} \end{bmatrix} = \begin{bmatrix} t_V & t_\gamma & t_\alpha & \sim 0 \\ p_V & \sim 0 & p_\alpha & 0 \\ -p_V & \sim 0 & -p_\alpha & 1 \\ m_V & 0 & m_\alpha & m_q \end{bmatrix} \cdot \begin{bmatrix} \Delta V/V \\ \gamma \\ \alpha \\ q \end{bmatrix} + \begin{bmatrix} t_\pi & \sim 0 \\ 0 & \sim 0 \\ 0 & \sim 0 \\ m_\pi & m_{\delta q} \end{bmatrix} \cdot \begin{bmatrix} \Delta\pi \\ \delta q \end{bmatrix}$$

La seule différence entre avion naturel et avion asservi résidera dans la valeur des termes m_V, m_α, et m_q : les asservissements élémentaires permettant de moduler à volonté ces coefficients.

On aura par exemple :

$$m'_V = m_V + m_{\delta q} \cdot K_V$$
$$m'_\alpha = m_\alpha + m_{\delta q} \cdot K_\alpha$$
$$m'_q = m_q + m_{\delta q} \cdot K_q$$

Conservant une structure "naturelle", les points caractéristiques F_V, F_α, F_q, pourront être redéfinis pour l'avion asservi, ceci permettant de rattacher la valeur des asservissements à des réalités "aérodynamiques".

A partir des formulations approchées donnant la position de ces points fondamentaux en fonction des caractéristiques aérodynamiques principales d'un avion de type AIRBUS (cf. planche 2), pour illustrer cette situation, il est fourni sur la planche 3 l'influence des asservissements sur la position de ces points.

3.2 Asservissements élaborés

Nous verrons plus loin que les asservissements élémentaires, s'ils permettent de stabiliser l'oscillation d'incidence, d'uniformiser les efforts par g, de reconstituer une stabilité statique, en aucun cas ne jouent le rôle d'autotrim. C'est au pilote lui-même de trouver son braquage d'équilibre, celui-ci étant trimmé afin de piloter autour d'un zéro d'effort.

Or, les variations du braquage d'équilibre sont fréquentes. Ainsi, chaque fois que la vitesse, la configuration, la poussée, la valeur du vent évoluent, l'intervention du pilote pour maintenir la pente de vol à effort moyen ou manche nul est nécessaire.

Afin d'éviter ces actions pilote, on pourra automatiser cette fonction par des asservissements à la pente, du type trim automatique ou maintien de pente.

321 Trim automatique

Cet asservissement sommaire à la pente consiste à trouver la position d'équilibre de la commande en sortie d'un intégrateur dont l'entrée est le facteur de charge normal n_z compensé de la pesanteur, et reconstitué à partir de la détection de l'évolution de la pente (cf. planche 4). A toute variation de la pente correspondra une modification de la position moyenne de la commande, de façon à assurer un vol à pente constante.

L'intégrateur permet au pilote de ne disposer que d'une seule commande de profondeur dont le zéro d'effort correspond toujours à l'équilibre, quelle que soit la pente de la trajectoire choisie. La sortie de l'intégrateur joue seulement le rôle de trim automatique, et non celui de maintien de pente, puisque pour toute perturbation, l'avion se restabilise sur une pente autre que celle initiale.

Afin de remédier à cette situation, un asservissement permettant de maintenir la pente sélectée, pourra être rajouté. La commande ainsi réalisée sera du type maintien de pente.

322 Asservissement du type maintien de pente

A partir :

- d'un asservissement assurant un trim automatique,
- d'un asservissement permettant de maintenir la pente sélectée,

on pourra réaliser une commande dont le principe est le suivant (voir planche 5) :

- sans action du pilote, l'avion suit une pente de référence imposée qui est celle acquise par l'avion au moment où le pilote lâche la commande,
- sur action du pilote sur le manche, par suppression de l'asservissement $K_\gamma \cdot (\gamma - \gamma_{\text{commandée}})$, l'avion évolue suivant une commande de type trim automatique.

4. ETUDE DES ASSERVISSEMENTS ELEMENTAIRES

Nous avons vu précédemment que les asservissements élémentaires permettent de déplacer les points fondamentaux (F_V, F_α, F_q) de l'avion. Cette propriété sera utilisée pour définir la valeur des asservissements en évaluant le positionnement optimum de ces points fondamentaux en fonction :

- des caractéristiques statiques (braquages par g et par V),
- des caractéristiques dynamiques (temps de réponse, amortissement des modes),

que l'on désire.

4.1 Caractéristiques statiques

Les caractéristiques statiques, à savoir les braquages par g et les braquages par V (relation traduisant la stabilité statique) sont directement proportionnelles respectivement aux distances FqXG (marge de manoeuvre) et FvXG (marge statique). Nous nous proposons de rechercher ci-après le positionnement optimal des points Fq et Fv.

41_1 Positionnement du point de manoeuvre

Pour un avion de transport subsonique, compte tenu des différentes possibilités de chargement de l'avion (passagers, fret, carburant), la position du centre de gravité peut évoluer d'un vol à l'autre dans des proportions non négligeables. Il en résulte une variation de braquages par g entre centrage avant et centrage arrière d'autant plus grande que le point de manoeuvre est avant.

On peut espérer réduire la dispersion des efforts par g entre centrage avant et centrage arrière en utilisant une sensation artificielle d'efforts dont la raideur est modulable avec le centrage de l'avion. Pour un avion AIRBUS, en croisière, pour des braquages par g variant avec le centrage dans un rapport de 1 à 2, on peut espérer des efforts par g quasiment identiques.

Dans le cas d'un avion à stabilité réduite, la plage de centrage est décalée vers l'arrière, sans pour autant que le point de manoeuvre de l'avion naturel soit modifié. Par exemple, pour un recul de la plage de centrage de 10 % de CAM, les marges de manoeuvres, donc les braquages par g entre centrages avant et arrière, varient dans un rapport de 1 à 4.5, rapport difficilement résorbable par une sensation artificielle d'efforts.

Donc, au moins pour des problèmes d'efforts par g, il sera nécessaire de reculer le point de manoeuvre de l'avion, ceci pouvant être obtenu par un asservissement de α , q ou nz. Un recul de 10 % de ce point permettra de retrouver la situation de l'avion AIRBUS (Cf. planche 6). Une solution plus astucieuse sera de moduler le déplacement du point de manoeuvre avec le centrage, le recul de 10 % n'étant atteint que pour le centrage maxi arrière. Cette solution permet avec le minimum d'asservissement, d'homogénéiser à la fois les braquages et les efforts par g (Cf. planche 6).

Une autre solution pour uniformiser braquages et efforts par g est tout naturellement de translater de façon importante le point de manoeuvre. Cette solution n'est toutefois pas sans risque, car :

- l'autorité du stabilisateur devient importante, d'où des problèmes découlant d'un éventuel embarquement,
- les gouvernes peuvent s'agiter en turbulence,
- la constante de temps des servocommandes impose une limite à la valeur de l'asservissement.

41_2 Positionnement du foyer en vitesse

Compte tenu de la situation actuelle des règlements de certification, il est impératif de justifier d'un effort par V respectant le critère de la livre par 6 kts.

Dans le cas d'avion non asservi, pour certaines positions du foyer en vitesse, le respect de cette exigence nécessitant l'augmentation des efforts par V peut conduire à durcir plus que nécessaire les efforts par g, compte tenu de la dépendance entre les efforts par g et par V explicitée par la relation suivante :

$$\frac{\text{Effort}/\Delta V/V}{\text{Effort}/n_z} \sim 2\,\frac{Fv\ XG}{Fq\ XG} \qquad (1)$$

C'est, par exemple, le cas de l'avion AIRBUS aux faibles Mach. Toutefois, aux Mach élevés, la modification de la raideur en profondeur n'est pas suffisante pour respecter ce critère. En effet, l'évolution du Cm_0 place Fv trop en avant, de sorte qu'il est illusoire de vouloir obtenir un effort par V d'au moins 1 livre par 6 kts sans asservissement. On sera amené à utiliser un Mach trim qui repositionnera le foyer en vitesse et qui permettra l'obtention d'efforts par g corrects (Cf. planche 7).

Cas de l'avion à stabilité réduite

Pour ce type d'avion, le foyer en vitesse de l'avion naturel peut être situé en tous points du domaine de vol en avant de la limite arrière du centrage. Il semble donc impensable de vouloir rétablir une situation correcte (effort au manche raisonnable et respect de la livre par 6 kts) sans l'aide d'asservissements.

On pourra profiter de l'utilisation de ces asservissements, qui, rappelons-le, peuvent être modulés avec le centrage de l'avion, pour réduire les efforts par g.

Avec un réglage de l'ordre de 30 lb par g, une condition simple pour positionner Fv peut s'expliciter sous la forme suivante :

$$\frac{Fv\ XG}{Fq\ XG} \geqslant \frac{Vc\ (kts)}{360}$$

41_3 Solution minimale pour le positionnement de Fq et Fv

Nous venons de voir que le respect des exigences attachées aux caractéristiques statiques conduit à définir une solution minimale pour le positionnement de Fq et Fv.

Cette solution minimale pour un avion à stabilité réduite et dans un cas de vol à 250 kts est présentée planche 8, en tenant compte :

- d'un positionnement du point de manoeuvre permettant d'homogénéiser les efforts et les braquages par g avec le centrage (Cf. § 4.1.1)
- d'un positionnement du foyer en vitesse assurant :
  . le respect de la livre par 6 kts,
  . un réglage d'efforts par g à $\sim$ 30 lb/g.

4.2 Caractéristiques dynamiques

421 Critère : temps de réponse

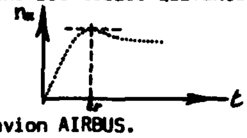
Les temps de réponse (ou périodes) de l'oscillation d'incidence et de la phugoïde dépendent respectivement de la position des points fondamentaux Fq et Fv.

Ces deux points ont été positionnés au chapitre précédent en fonction des caractéristiques statiques, *dans une solution minimale. L'étude* et l'analyse des caractéristiques dynamiques permettra au contraire de définir une solution maximale.

421.1 Positionnement maximal du point de manoeuvre

Il est tentant du point de vue du mouvement dynamique de l'avion d'avoir une forte marge de manoeuvre, afin de diminuer les temps de réponse de cet avion. Rappelons à cet effet que sur un échelon de la commande en profondeur, le facteur de charge est obtenu dans les délais suivants :

$$tr \sim 3{,}5\left(\frac{g}{\rho y^2}\cdot\frac{Cz_\alpha}{Cz}\cdot\overline{Fq\ X_G}\right)^{-\frac{1}{2}}$$



Cette relation est illustrée sur la planche 9 pour l'avion AIRBUS.

D'aucuns trouveront préférable d'avoir un temps de réponse court et uniforme dans tout le domaine de vol. Ceci conduira à augmenter la marge de manoeuvre au fur et à mesure que la vitesse diminue. La recherche de temps de réponse courts, à basses vitesses, conduit à des marges de manoeuvres nécessitant des valeurs d'asservissement dont une limite maximale sera imposée par la valeur de la constante de temps (τsc) des servocommandes de profondeur.

Pour l'avion considéré, le temps de réponse de l'oscillation d'incidence est limité comme suit :

$$tr > 14.\tau\ sc$$

L'obtention d'un temps de réponse de 2,2 secondes imposera une constante de temps de servocommande de l'ordre de 0,16 seconde, valeur tout à fait compatible avec les organes de puissance équipant la profondeur de l'avion AIRBUS (τ sc = 0,125 s). Cette limite serait toutefois sévère pour une commande de profondeur utilisant le plan horizontal.

421.2 Positionnement du foyer en vitesse

Autant l'on recherche une réponse rapide en facteurs de charge, autant pour la phugoïde l'on désire une période aussi longue que possible, caractéristique d'une stabilité statique très modérée et dont le niveau minimum est imposé par le Certificateur.

De plus, il est préférable pour la facilité du pilotage de bien découpler phugoïde et oscillation d'incidence. On essayera de respecter la relation suivante entre les pulsations des 2 modes :

$$\omega_\varphi \leq \frac{1}{6{,}3}\cdot\omega_\alpha$$

avec :

$$\omega_\alpha^2 = \frac{g}{\rho_y^2}\cdot\frac{Cz_\alpha}{Cz}\cdot\overline{Fq\ X_G} \qquad \omega_\varphi^2 = 2\cdot\left(\frac{g}{V}\right)^2\cdot\frac{\overline{Fv\ X_G}}{\overline{Fq\ X_G}}$$

d'où la relation approchée en fonction du temps de réponse :

$$\left(\frac{\overline{Fv\ X_G}}{\overline{Fq\ X_G}}\right)^{\frac{1}{2}} < \frac{V(m/s)}{25.tr}$$

Afin de respecter le niveau de stabilité statique réglementaire (Cf. § 4.1.2), on est conduit à diminuer le temps de réponse de l'avion de telle sorte que :

$$tr < 0{,}8\,\frac{V(m/s)}{Vc^{\frac{1}{2}}\ (kts)}$$

Le respect de cette condition n'apparaît limitatif qu'aux basses vitesses et impose par exemple à 125 kts une marge de manoeuvre minimale de 30 % de CAM (Cf. pl. 9).

421.3 Conclusions sur la position de Fq et Fv

La figure 9 définit un domaine dans lequel il est possible de positionner Fq. On note que ce domaine est restreint à grandes vitesses pour laisser place à des possibilités variées aux basses vitesses.

La position de Fv découle de Fq :

- par la condition de stabilité statique $\frac{\overline{Fv\ X_G}}{\overline{Fq\ X_G}} > \frac{Vc\ (kts)}{360}$
- par la condition sur le temps de réponse de l'avion $\frac{\overline{Fv\ X_G}}{\overline{Fq\ X_G}} < \left(\frac{V(m/s)}{25\cdot tr}\right)^2$

Le foyer en vitesse de l'avion naturel dépendant du niveau de poussée des moteurs, l'asservissement en vitesse devra donc être modulé avec le régime des moteurs, à moins que la double inégalité précédente soit toujours respectée pour la gamme des régimes moteurs à considérer. Une solution maximale de positionnement de ces points pour un avion à stabilité statique réduite est présentée planche 10.

4_{22} Amortissements

Les points fondamentaux Fq et Fv ayant été positionnés afin de régler au mieux les périodes de l'oscillation d'incidence et de la phugoïde, nous nous intéresserons ci-après aux amortissements de ces modes.

$4_{22.1}$ Oscillation d'incidence

Si, réglementairement, on exige un amortissement minimal, on s'attachera pour un avion asservi à le porter à un niveau optimal pour le confort du pilotage.

A marge de manoeuvre donnée (Fq positionné), le niveau d'amortissement souhaité sera obtenu par un positionnement judicieux du foyer en incidence ($F\alpha$).

La règle à respecter sera la suivante :

$$1 + \frac{\overline{Fq\,F_\alpha}/\ell}{\frac{1}{2}\,\rho\,(\rho\,y^2/\rho).\,\frac{S}{m}\,.\,C_{Z_\alpha}} = 2\,\xi_\alpha \left[\frac{\overline{Fq\,X_G}/\ell}{\frac{1}{2}\,\rho\,.\,\rho y^2/\rho\,.\,\frac{S}{m}\,.\,C_{Z_\alpha}}\right]^{\frac{1}{2}}$$

Dans le cas de l'avion AIRBUS cette relation illustrée sur la planche 11 permet de trouver la relation entre $FqF\alpha$ et FqX_G pour un amortissement donné de l'oscillation d'incidence.

Nous avons vu précédemment qu'en croisière la solution pratiquement unique pour le positionnement de Fq se situe autour de $\frac{Fq\ XG}{l}$ = 15 % CAM, ce qui impose un positionnement de $F\alpha$ tel que $\frac{Fq\ F\alpha}{l} \sim 8$ % CAM. On peut noter au passage que nous nous retrouvons dans la situation de l'avion AIRBUS au centrage arrière.

Dans le cas d'un avion à stabilité réduite, centré à 45 % de CAM, il convient de reculer le foyer en incidence et le point de manoeuvre de $\sim$ 10 % pour retrouver les conditions précédentes. Un asservissement au facteur de charge seul, suffit donc en croisière, sans besoin d'asservissement en q, puisque Fq recule avec $F\alpha$ (Cf. planche 12). Le fait de créer une marge de manoeuvre de 15 % par un asservissement en q seul, aboutirait d'ailleurs à ne pas régler correctement l'amortissement de l'oscillation d'incidence.

$4_{22.2}$ Phugoïde

La question de l'amortissement de la phugoïde se pose différemment. Pour bien l'amortir, il faudrait, soit réaliser un asservissement à la pente, ce qui sort du cadre des asservissements élémentaires, soit ne pas exiger une stabilité statique aussi forte.

En fait, on peut être assuré que la phugoïde sera convergente dans la mesure où l'on aura respecté les réglages précédents, bien que son amortissement reste très faible.

5. ETUDE DES ASSERVISSEMENTS ELABORES DU TYPE AUTOTRIM

Comme pour les asservissements élémentaires, nous allons nous intéresser dans ce paragraphe aux effets d'une commande de type autotrim sur les caractéristiques statiques puis dynamiques de l'avion.

5.1 Caractéristiques statiques

5_{11} Braquages et efforts par g

Tout ce qui a été dit dans le chapitre 4.1 concernant le positionnement de Fq pour les asservissements élémentaires reste valable. A savoir que l'on peut, comme précédemment, définir une solution dite minimale pour positionner Fq, permettant d'homogénéiser braquages et efforts par g, quel que soit le centrage de l'avion. Nous verrons toutefois d'après l'étude du dynamique, que cette solution minimale ne présente pas d'intérêt pour le positionnement optimum de Fq.

5_{12} Braquages et efforts par V

Dans le cas d'une loi de pilotage de type trim automatique, l'asservissement de la commande de profondeur au facteur de charge intégré introduit, par rapport à la structure de l'avion naturel (Cf. § 3.1), un accroissement en pente tel que :

$$m_\gamma = \frac{1}{\tau_{at}} \cdot \frac{\overline{m_\alpha}}{p_\alpha}$$

avec $\overline{m_\alpha}$: contribution de l'asservissement à n_z dans m_α

τ_{at} : *constante de temps de l'intégration d'autotrim.*

La matrice des coefficients linéarisés de l'avion se présente sous la forme ci-dessous :

$$\begin{bmatrix} A \end{bmatrix} = \begin{bmatrix} tv & t_\gamma & t_\alpha & \sim 0 \\ pv & 0 & p_\alpha & \sim 0 \\ -pv & 0 & -p_\alpha & \sim 1 \\ mv & m_\gamma & m_\alpha & mq \end{bmatrix}$$

La stabilité statique de l'avion manche libre sera donnée par le signe du déterminant de (-A). Pour un avion à stabilité réduite, ce déterminant est négatif dans une grande partie du domaine de vol, l'instabilité étant due aux raisons suivantes :

- d'une part à pente libre, nous avons vu dans l'étude des asservissements élémentaires que l'avion à stabilité réduite n'est stable que dans une très faible partie du domaine de vol ;
- d'autre part à pente imposée, l'avion est instable en vitesse pour les vols à forts C_Z, au-delà de la finesse maximale.

Dans le cas présent ces deux effets se conjuguent puisque la pente de la trajectoire n'est pas tout à fait libre, sans être non plus complètement imposée.

L'analyse de l'expression du déterminant de (-A) permet de faire la part des deux phénomènes.
On peut exprimer le déterminant sous la forme suivante :

$$\det(-A) = t_\gamma \cdot \begin{vmatrix} p_v & p_\alpha \\ m_v & m_\alpha \end{vmatrix} + m_\gamma \cdot \begin{vmatrix} t_v & t_\alpha \\ p_v & p_\alpha \end{vmatrix}$$

Pour les asservissements élémentaires ($m_\gamma = 0$), nous avions défini le positionnement du foyer en vitesse à partir du 1er terme de ce déterminant :

$$t_\gamma \cdot \begin{vmatrix} p_v & p_\alpha \\ m_v & m_\alpha \end{vmatrix} = 2 \cdot \left(\frac{g}{V}\right)^2 \cdot \left(\frac{g}{\rho_y{}^2} \cdot \frac{C_{Z\alpha}}{C_Z} \cdot \overline{F_v X_G}\right)$$

Par extension pour $m_\gamma \neq 0$, nous pouvons écrire :

$$\det(-A) = 2\left(\frac{g}{V}\right)^2 \cdot \left(\frac{g}{\rho_y{}^2} \cdot \frac{C_{Z\alpha}}{C_Z}\right) \cdot \left(\overline{F_v X_G} + \overline{\Delta F_v}\right)$$

le terme ΔF_v chiffrant la part de l'effet "second régime". Il est possible par un asservissement à la vitesse de restituer un effort par V répondant aux exigences réglementaires (Cf. planche 13, mais avec l'inconvénient pour le pilote de devoir se retrimmer à chaque changement de vitesse. Ce type de loi a d'ailleurs été essayé sur simulateur de vol. Il va sans dire que les réglages les plus agréables pour le pilotage correspondaient aux valeurs les plus faibles pour l'asservissement.

5.2 Caractéristiques dynamiques

5.2.1 Temps de réponse

Par rapport à un système utilisant des asservissements élémentaires, compte tenu de la présence de l'intégrateur, le temps de réponse d'un système à asservissements élaborés est augmenté à marge de manoeuvre donnée.

Pour l'avion à stabilité réduite, à partir des résultats de la planche 14, on constate qu'il faut s'autoriser des marges de manoeuvres quatre fois plus grandes que celles évaluées pour les asservissements élémentaires, et ce pour un même temps de réponse.

Dans la recherche de temps de réponse courts, nécessitant des marges de manoeuvres importantes (surtout aux basses vitesses), il sera impératif de prendre garde aux limitations imposées par la constante de temps des servocommandes.

On devra respecter la condition suivante entre tr et τ_{sc} : $tr \sim 25 \cdot \tau_{sc}$, ce qui nous conduira à disposer de servocommandes plus rapides que pour les asservissements élémentaires.

Quant à la constante de temps de l'intégrateur d'autotrim (τ_{AT}), elle est liée au temps de réponse par la relation : $tr \sim 3{,}3 \cdot \tau_{AT}$. Cette constante de temps évoluera de :
- 1 seconde à grandes vitesses ($tr \sim 3s$) à
- 1,6 seconde aux basses vitesses ($tr \sim 5s$).

Nota : On se contente de ce temps de réponse aux basses vitesses afin de limiter l'autorité des asservissements.

Compte tenu des marges de manoeuvre envisagées, on conçoit aisément :
- que les réponses de l'avion à la commande pilote sont quasiment insensibles aux variations de centrage,
- que le découplage entre oscillation d'incidence et phugoïde (positionnement de Fv par rapport à Fq) est satisfait.

5.2.2 Amortissement

Les résultats de l'étude des amortissements dans le cas des asservissements élémentaires, à savoir :
- la relation définissant la position respective de F_α par rapport à Fq et X_G (Cf. § 4.2.2.1) dans le but d'amortir convenablement l'oscillation d'incidence,
- les remarques faites au sujet du réglage de l'amortissement de la phugoïde,

s'appliquent à ce type de commande.

6. ETUDE DES ASSERVISSEMENTS DE TYPE MAINTIEN DE PENTE

Nous avons vu précédemment que pour ce type de commande :
- sans action pilote, l'avion suit une pente de référence imposée,
- sur une action du pilote l'avion évolue en répondant conformément à une commande de type trim automatique.

De cette situation, il résulte que :
- les réglages évalués pour la commande dite de trim automatique sont valables (hormis les problèmes de stabilité statique),
- seul le réglage de l'asservissement en pente reste à définir.

Cet asservissement à la pente sera réglé tel que la pulsation de coupure de la dynamique de maintien de pente (w_γ) soit 2,5 fois plus lente que celle de la dynamique de la boucle de base (w_{nz}), soit :

$$K_\gamma = \frac{V}{g} \cdot \frac{0{,}63}{tr} \quad (K_\gamma \text{ en g/rd}).$$

6.1 Protection de domaine et stabilité statique

En l'absence d'action du pilote, l'avion est en pur maintien de pente. Il sera stable ou non en vitesse selon qu'il volera au 1er ou 2ème régime. Or, si l'on ajoute un terme de stabilité en vitesse, on annule la fonction "maintien de pente". Il serait donc aberrant, pour ce type de commande, de l'introduire dans le domaine normal de vol.

Cependant, compte tenu du fait qu'une action pilote conduisant involontairement à une variation de pente, risque d'amener l'avion à une vitesse en dehors du domaine autorisé, et cela même si l'avion est stable, on introduira à la frontière du domaine normal de vol, un terme de stabilité en vitesse en le réglant de façon à ce qu'il assure un maintien de vitesse.

On adoptera : $Kv = \frac{K\gamma^2}{2,5.V}$ (Kv en g/m/sec), ce qui correspond à un réglage tel que la pulsation de coupure de la dynamique du maintien de vitesse soit 2,5 fois plus lente que celle du maintien de pente.

Dans ce maintien de vitesse, l'asservissement à la pente est maintenu avec le même réglage pour servir de terme d'amortissement. Dans ce rôle on l'affectera donc d'un filtre passe-haut, pour laisser au maintien de vitesse la liberté d'amener l'avion à la pente adéquate (Cf. planche 15).

7. ADAPTATION DE CES TYPES DE COMMANDES DE VOL A UN AVION CIVIL DE TRANSPORT

Les différents asservissements présentés dans cette note permettent suivant leur niveau d'autorité et de complexité de restituer des qualités correctes de pilotage quelles que soient les marges statiques et de manoeuvres de l'avion sans asservissements.

Toutefois, bien que cela soit à la limite du sujet de cette communication, l'on ne devra pas perdre de vue que l'on peut se trouver en vol dans des configurations de pannes, de sorte que l'évaluation du pilotage dans des situations dégradées (perte de calculateurs de commandes de vol par exemple), devra être prise en considération pour les définitions de l'avion et de l'architecture de ses commandes de vol.

7.1 Cas de commandes de vol mécaniques

Un avion à stabilité réduite diffère des avions classiques par le fait que dans une grande partie du domaine de vol et de la plage de centrage autorisée, l'avion naturel présente une certaine instabilité longitudinale.

Suivant que le centrage avion pour un chargement donné :

- n'évolue que très faiblement en vol avec la consommation du carburant,
- peut évoluer en vol par le transfert du carburant des réservoirs de voilure vers des réservoirs situés dans la partie arrière de l'avion, et vice versa,

le niveau d'instabilité accepté après perte des asservissements sera tout naturellement différent.

Ainsi, dans le cas où le centrage reste fixe, le niveau d'instabilité toléré devra permettre la poursuite du vol et l'atterrissage, alors que dans le cas du centrage évolutif, le niveau d'instabilité toléré devra être compatible avec le maintien en vol de l'avion en attendant que, par transfert du carburant, le centrage revienne vers une position où l'instabilité de l'avion est plus faible, voire nulle.

Des essais conduits sur simulateur de vol et sur un avion AIRBUS ont montré que la différence du niveau d'instabilité accepté dans les deux types d'avion peut se traduire par une variation de 5 % de centrage. L'on devra donc assurer pour chaque définition avion la compatibilité conséquence avion - probabilité d'occurrence de panne, ces conditions définissant le centrage limite arrière acceptable.

7.2 Cas de commandes de vol électriques

Bien qu'un avion à stabilité réduite peut parfaitement se concevoir avec des commandes de vol mécaniques, l'évolution de la technologie conduit à essayer de les remplacer par des commandes électriques, et cela pour de multiples raisons, telles que :

- gain de masse,
- diminution du prix de revient,
- amélioration de qualité de pilotage (seuils et frottements réduits, braquages par g homogènes, diminution des problèmes liés au déroulement de trim ou centrage avant...),
- meilleure situation en turbulence.

Si, pour un avion à stabilité réduite, l'on craignait dans le cas des commandes de vol mécaniques la perte des asservissements, la perte de la commande et ses conséquences sur la définition des systèmes étant identiques à celles des avions de conception classique, dans le cas des commandes de vol électriques on devra tenir compte, dans la définition de l'architecture des commandes de l'avion, de l'éventualité de perdre momentanément ou définitivement les commandes principales.

Une solution possible, pour résoudre cette situation de panne, sera par exemple de baser l'architecture des commandes de vol sur un pilotage secours avec des commandes mécaniques :

- par le trim de profondeur (le plan horizontal réglable reste commandé mécaniquement),
- par la gouverne de direction (elle reste en commande mécanique) dans le cas où la commande de gauchissement est électrique.

Des situations de panne moins dégradées mais correspondant à des probabilités d'occurrence moins faibles seront également à prendre en considération. La ségrégation des divers circuits et la dispersion des différentes fonctions et asservissements dans plusieurs calculateurs permettront d'assurer la correspondance : probabilité d'occurrence de panne et conséquences avion.

8. PERSPECTIVES

Associant :

- gain en consommation par recul modéré du centrage en vol, sans pénalisation sur les conditions de chargement, d'une part,
- amélioration des qualités de pilotage, diminution du coût et de la masse de l'avion par utilisation de commandes de vol électriques, sans pour autant réduire le niveau de sécurité, celui-ci étant assuré par

un pilotage secours avec des commandes mécaniques sur certaines gouvernes, d'autre part, cette nouvelle génération d'avions devrait voler dans les prochaines années. L'expérience qui sera acquise au cours de milliers d'heures de vol sur ce type d'aéronef devra permettre de définir l'avion subsonique de transport civil du futur, celui-ci étant un avion volant au centrage optimum du point de vue de la consommation, avec des commandes de vol entièrement électriques et sans le secours de nos vétustes câbles, guignols et poulies.

NOTATIONS

V	Vitesse de l'avion
γ	Pente de la trajectoire
α	Incidence
q	Vitesse de tangage
n_z	Facteur de charge vertical
C_Z	Coefficient de portance, référencé à S
$C_{Z\alpha}$	Gradient du coefficient de portance, référencé à S
Π	Poussée des réacteurs
δq	Braquage de la gouverne de profondeur
ρ_y	Rayon de giration en tangage
X_G	Centre de gravité
F_α	Foyer en incidence
F_v	Foyer en vitesse
F_q	Point de manoeuvre
t_i	Coefficients de l'équation linéarisée de traînée
P_i	Coefficients de l'équation linéarisée de portance
m_i	Coefficients de l'équation linéarisée de tangage
S	Surface de référence de la voilure principale
S_H	Surface de référence de l'empennage horizontal
l_H	Distance voilure (25 % CAM) - empennage horizontal (25 % CAM)
l	Corde aérodynamique de référence de la voilure
$C_{Z\alpha H}$	Gradient du coefficient de portance de l'empennage horizontal, référencé à S_H
m	Masse de l'avion
ρ	Densité volumique de l'air
M	Mach
C_{m_0}	Coefficient de moment de tangage à portance nulle
l_π	Distance verticale des moteurs au centre de gravité
C_π	Coefficient de poussée des moteurs, référencé à S
$C_{Z\delta q}$	Gradient du coefficient de portance de l'empennage par rapport à δq, référencé à S_H
g	Accélération de la pesanteur
C_X	Coefficient de traînée, référencé à S
t_r	Temps de réponse
τ_{sc}	Constante de temps des servocommandes
τ_{at}	Constante de temps de l'autotrim
w_φ	Pulsation de l'oscillation de la phugoïde
w_α	Pulsation de l'oscillation de l'incidence
V_C	Vitesse conventionnelle ou corrigée
$(-A)$	Déterminant caractéristique des équations linéarisées de l'avion
A	Matrice des coefficients linéarisés ti, pi, mi
ξ_α	Amortissement de l'oscillation d'incidence
ξ_φ	Amortissement de la phugoïde

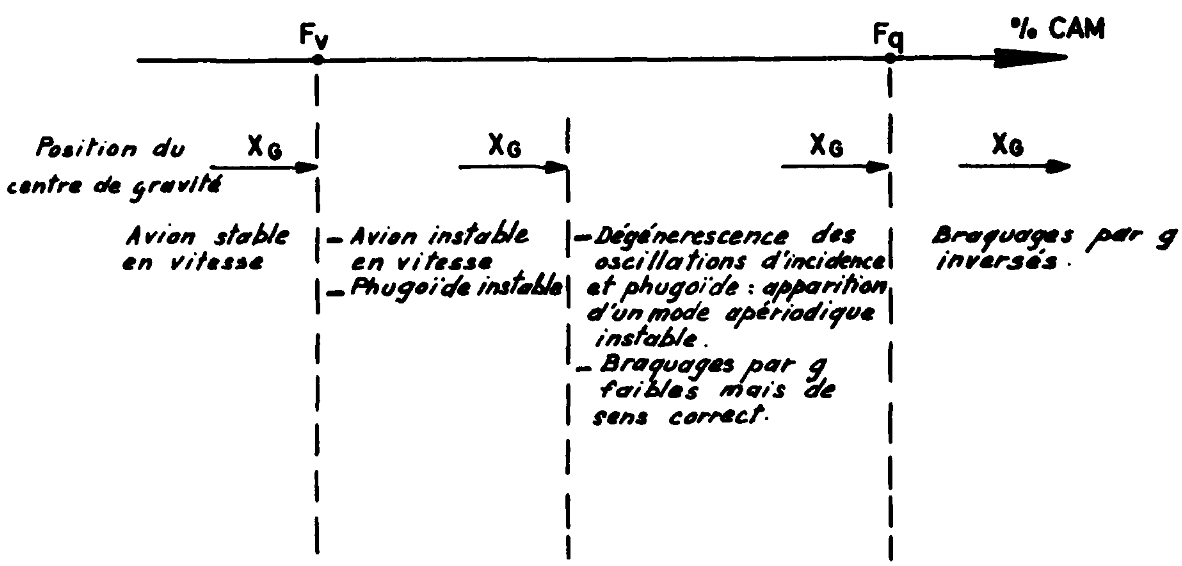


Planche : 1. Correspondance entre phénomènes aérodynamiques et positionnement des points caractéristiques.

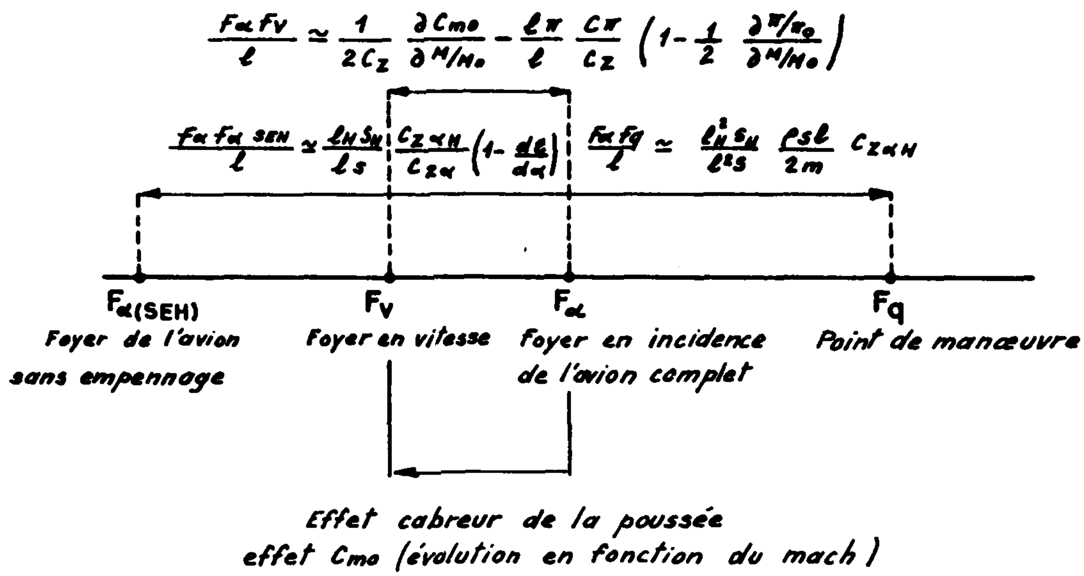


Planche: 2. Points caractéristiques de l'avion naturel

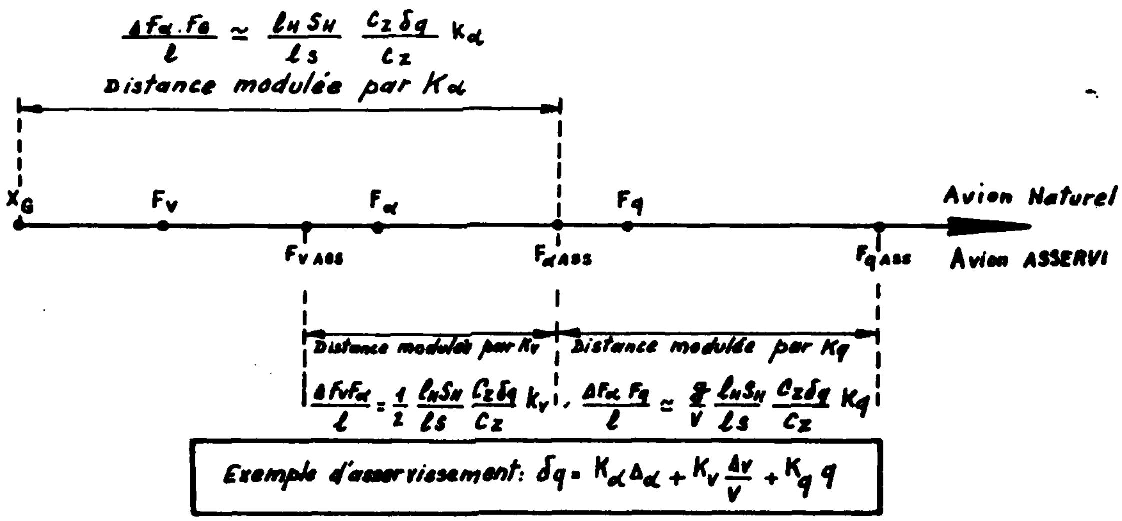


Planche: 3. Influence des asservissements sur la position des points caractéristiques.

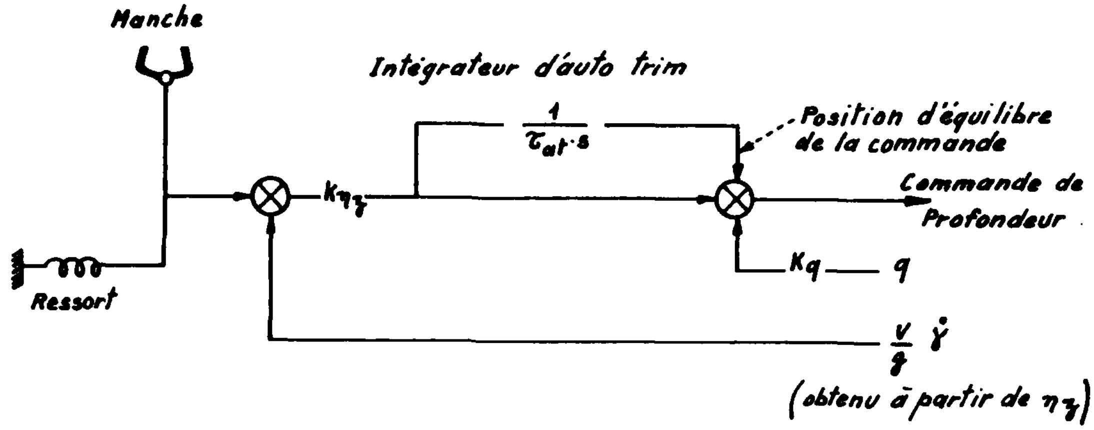
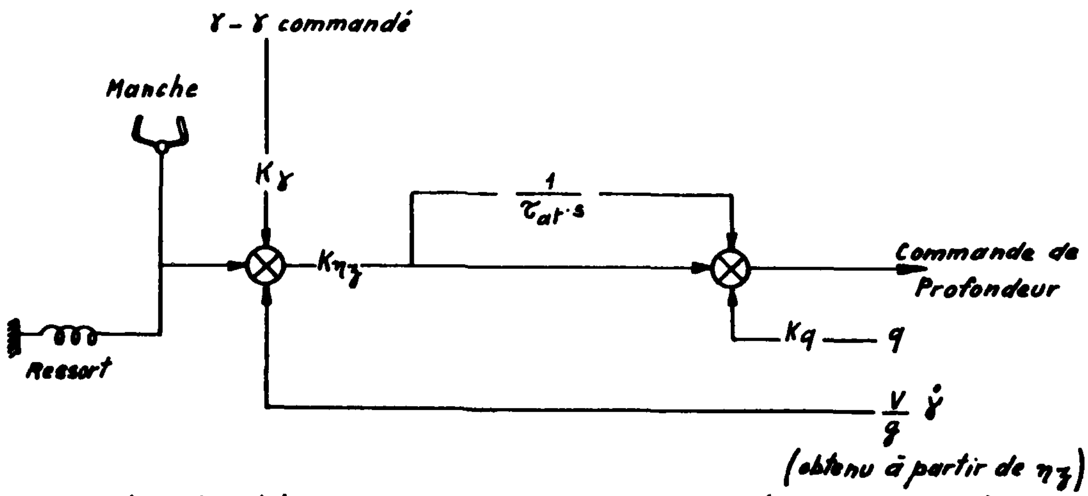


Planche : 4. Commande de type "Trim Automatique"



Nota : Sur action du pilote, suppression de l'asservissement ($\gamma - \gamma$ commandé) K_γ

Planche : 5. Commande de type "maintien de pente"

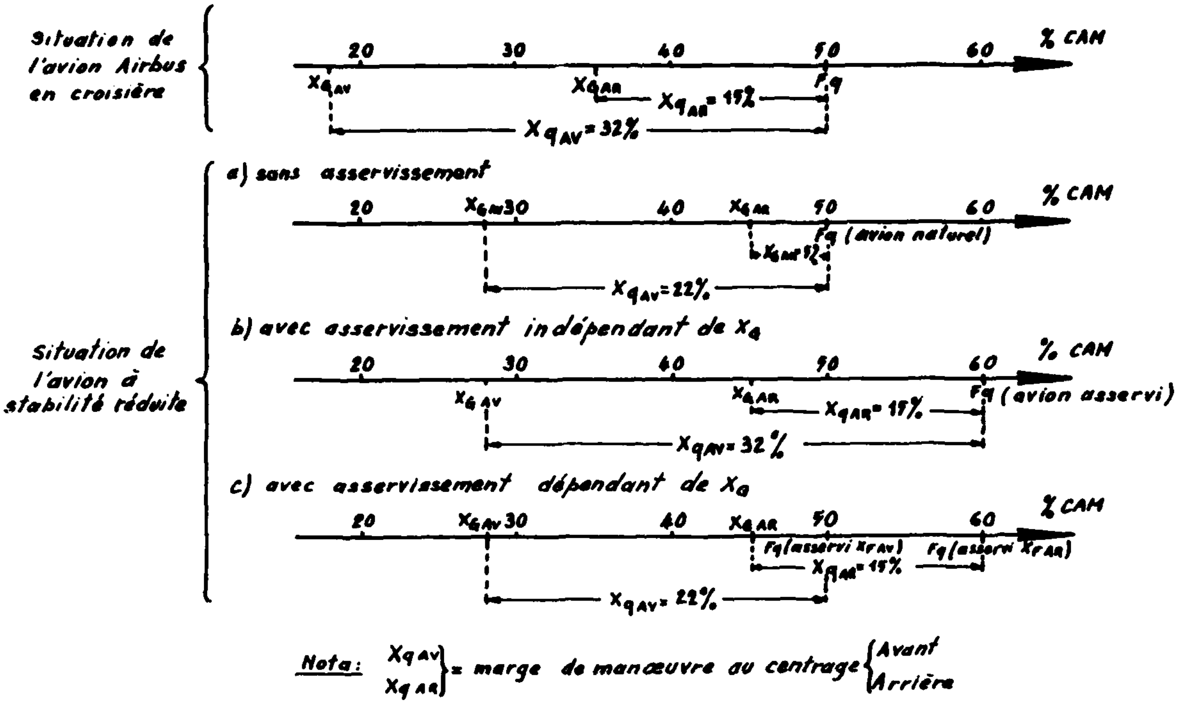


Planche : 6. Positionnement minimal du point de manœuvre

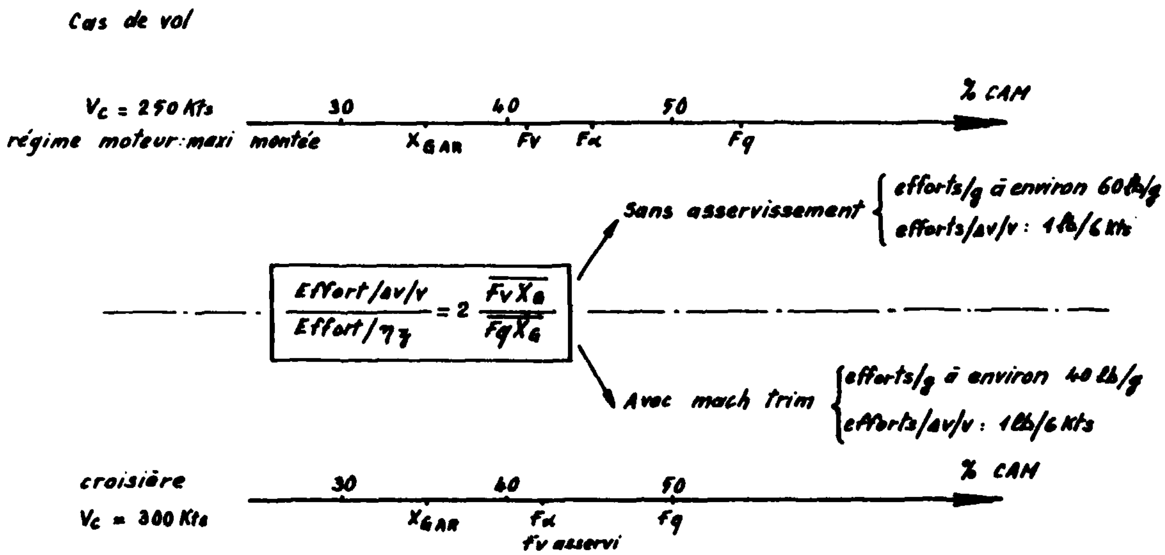
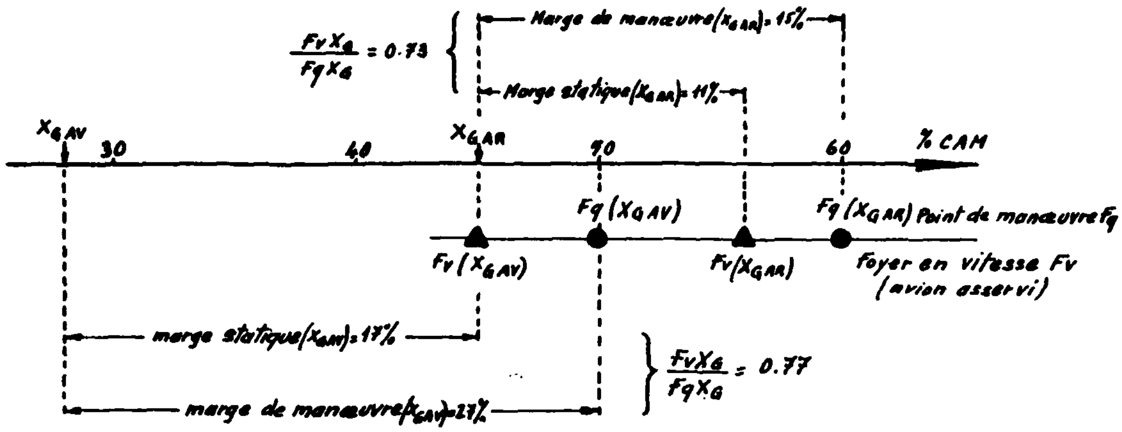


Planche : 7. Situation de l'avion Airbus vis à vis des efforts/g et efforts/v au centrage arrière.



Condition sur le positionnement de Fv avec un réglage de 30 lb/g :

$$\frac{Fv\,X_G}{Fq\,X_G} \geqslant \frac{V_c\ (Kts)}{360}$$

Planche: 8. Positionnements minimaux des points Fv et Fq pour un avion à stabilité réduite à Vc = 250 kts.

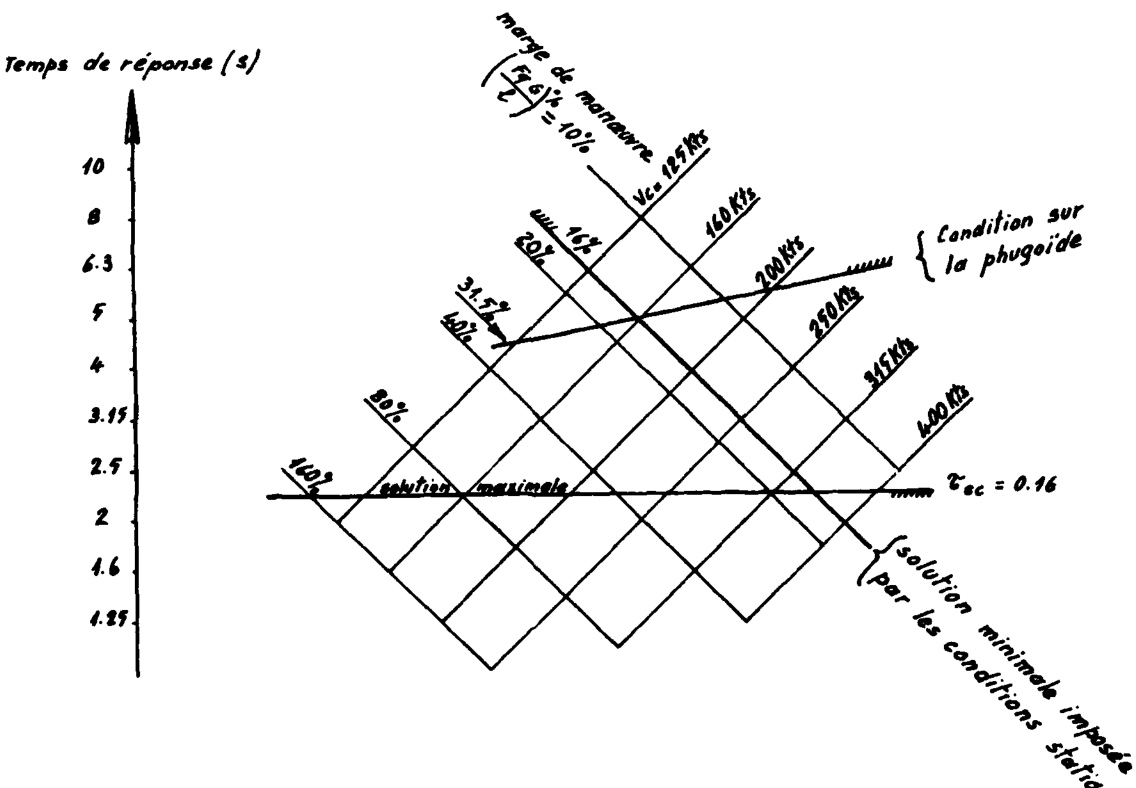


Planche: 9. Marges de manœuvre admissibles.

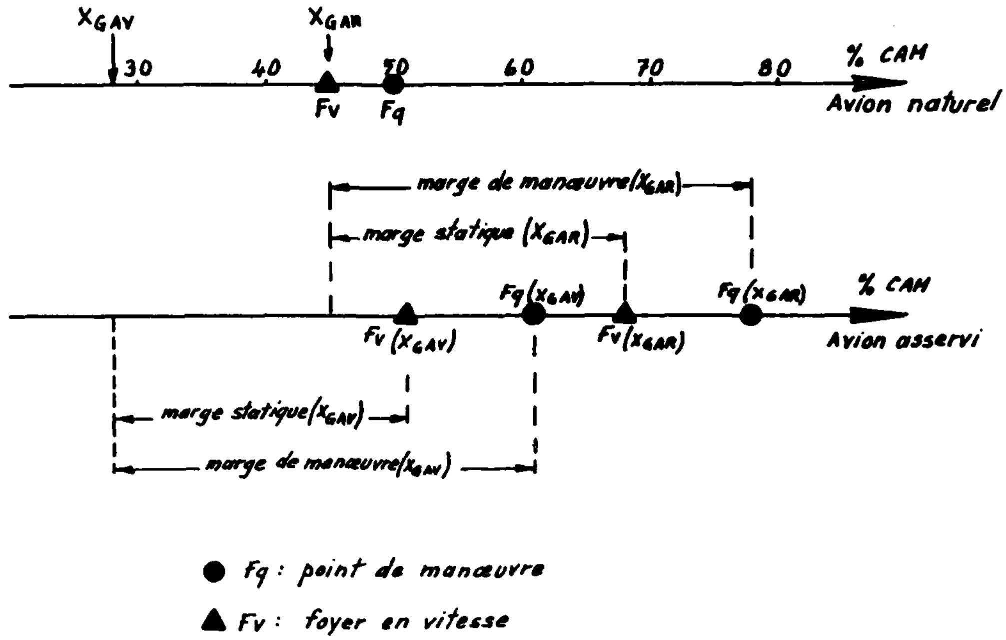


Planche:10. Positionnements maximaux des points Fv et Fq pour un avion à stabilité réduite à Vc= 250 kts.

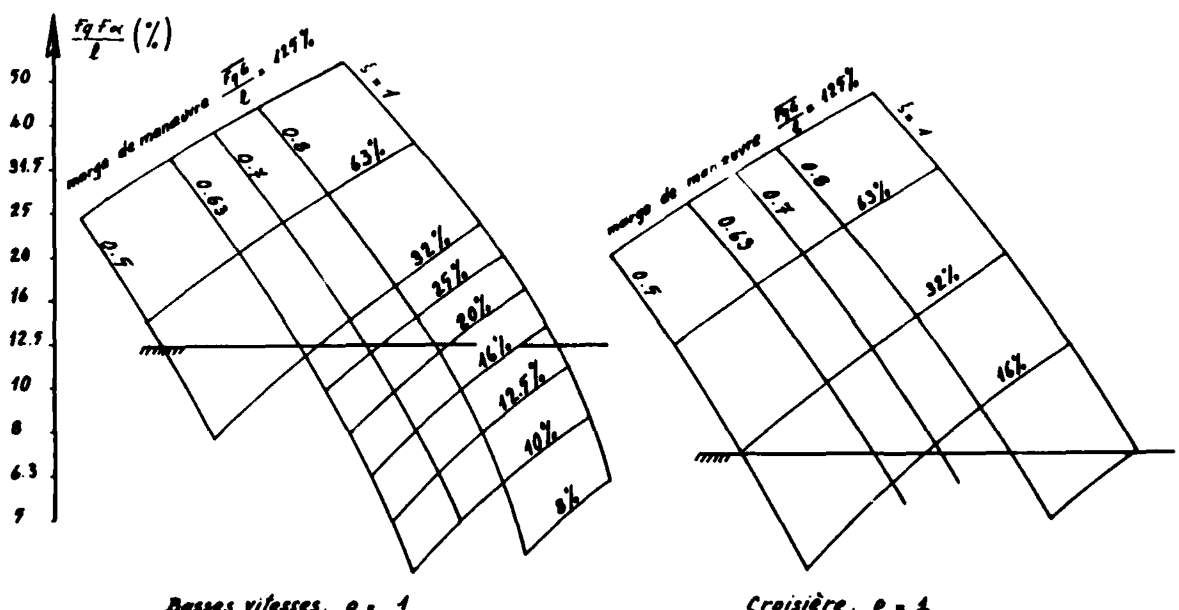


Planche:11. Position du foyer en incidence en fonction de la marge de manœuvre : réglage de l'amortissement de l'oscillation d'incidence.

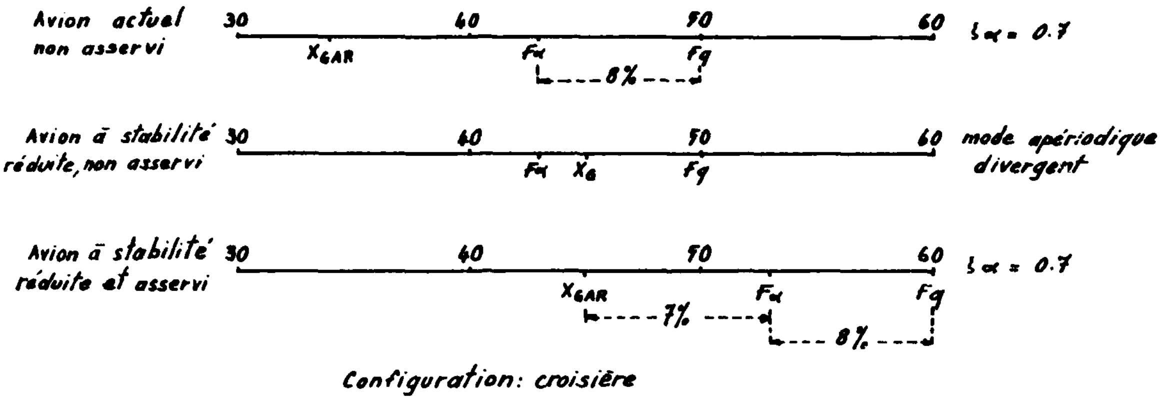


Planche:12. Positionnement du foyer en incidence.
Réglage de l'amortissement de l'oscillation d'incidence
(à Fq positionné).

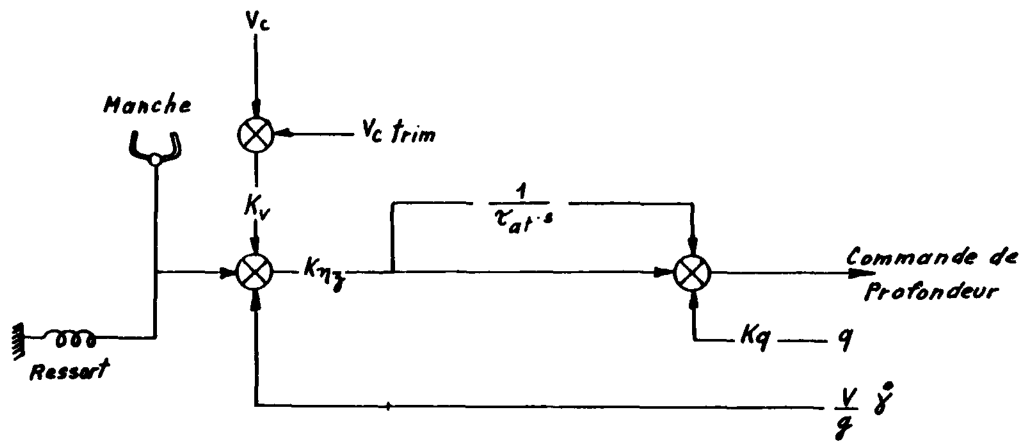


Planche:13. Commande de type "Auto trim et Stabilité Statique".

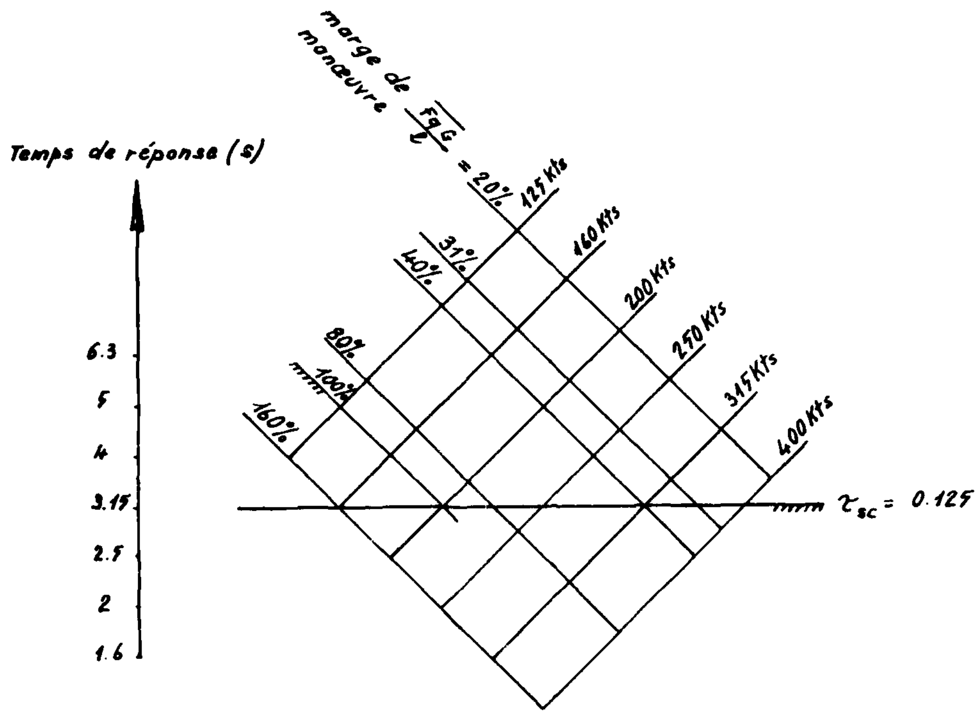


Planche: 14. Temps de réponse avion pour une commande de type auto trim.

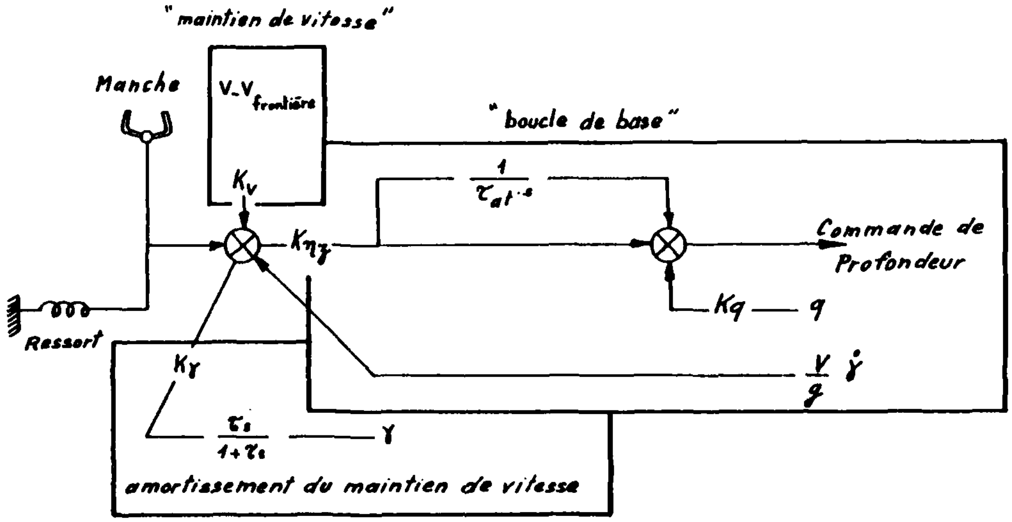


Planche: 15. Protection de domaine dans le cas de commande type "maintien de pente"

DEVELOPMENT OF HANDLING QUALITIES TESTING IN THE 70'S
A NEW DIRECTION

by

B. Lyle Schofield, Chief
Flight Test Technology Branch
Air Force Flight Test Center
Edwards Air Force Base, California, 93523

Thomas R. Twisdale, Aerospace Engineer
Flight Test Technology Branch
Air Force Flight Test Center
Edwards Air Force Base, California, 93523

Dr. **William G. Kitto**, Mathematician/Programmer
Software Development Branch
Air Force Flight Test Center
Edwards Air Force Base, California, 93523

Tice A. Ashurst, Mathematician/Programmer
(Formerly) Software Development Branch
Air Force Flight Test Center
Edwards Air Force Base, California, 93523

ABSTRACT

During the 1970's, the AFFTC (Air Force Flight Test Center) at Edwards Air Force Base began taking a significant new approach to handling qualities flight testing. New flight test and analysis techniques were developed and implemented. These techniques used pilot-in-the-loop, mission oriented tests to collect, analyze, and evaluate handling qualities data. These test and analysis techniques have had a profoundly favorable effect on the quality of test results obtained at AFFTC and on our ability to identify and evaluate handling qualities problems. We believe the development of these new test and analysis techniques was the pivotal development of the 1970's in the field of handling qualities flight testing.

Before 1971 the AFFTC relied very heavily on open-loop handling qualities testing. The data from these tests were compared with the flying qualities specification, MIL-F-8785, to determine whether or not the handling qualities were satisfactory. Limited pilot-in-the-loop handling qualities testing was done, but not in a systematic or repeatable way. Pilot compensation (adaptation) made it very difficult to interpret the quantitative and qualitative data that were produced.

In 1971 AFFTC began developing the pilot-in-the-loop, mission oriented handling qualities test and analysis techniques which were eventually named SIFT (System Identification From Tracking). SIFT includes specialized flight test techniques for collecting data and relatively sophisticated frequency domain analysis techniques for data reduction and evaluation.

The SIFT data collection techniques currently include several specialized and carefully controlled pilot-in-the-loop, mission oriented tasks such as precision air-to-air tracking, formation flying, air refueling, and approach and landing. Specialized piloting techniques were developed for these tasks in order to minimize the effect of pilot compensation (adaptation) and to increase the likelihood that handling qualities deficiencies are recognized and evaluated by the pilot. The careful controls placed on the task together with the specialized piloting techniques result in quantitative and qualitative test data which are repeatable.

The SIFT data analysis techniques include the use of spectral estimation methods to identify linear frequency response transfer functions of various systems. The "system" for which these results are obtained may be the entire airplane (airplane response to pilot input), or some smaller part of the whole airplane. For example, system transfer functions for the flight control system (control surface response to pilot input), or for pilot/airplane interfaces such as a head-up display may be identified.

The frequency response data may be used for analyzing handling qualities in terms of such recently developed criteria as equivalent systems, Neal-Smith, Ralph Smith, and bandwidth.

The quantitative frequency response data and the various criteria comparison results may be correlated with the qualitative pilot comments to provide significant insight into handling qualities characteristics. Because all of the data were obtained during the same pilot-in-the-loop, mission oriented manuevers, the correlation of qualitative and quantitative results is especially valuable.

An overview of the evolution of handling qualities testing at AFFTC is presented, with emphasis on the significant changes which occurred during the 1970's. The SIFT handling qualities test and evaluation techniques are discussed, but without going into great detail regarding the analysis techniques. Examples of SIFT test results are presented and discussed.

EARLY (HANDLING QUALITIES?) TESTING AT MUROC

If you believe "war stories", flight testing in the early days at Edwards was more fun, less structured, more relaxed, and more dangerous: in a word, wilder. (Reference 1 gives some additional insight.) That was in the days before Edwards was called "Edwards" and before it was a Flight Test Center. At that time Edwards was called Muroc Field and it was just a lonely flight test outpost of Wright Field.

Mr. Don Button, recently retired from Edwards, recounted the following story for us. As a young soldier during World War II, Don was stationed at Muroc, assigned to the P-59 test program. (The P-59 was America's first jet fighter.) One of Don's duties was to take data during the test flights, a duty which was shared with three or four other fellows. To accommodate Don and the other human data recorders a hole was cut into the nose of the P-59 just in front of the pilot's windscreen and a seat was installed where the guns were normally located. An oxygen system and an intercom were installed so that they could breathe and talk with the pilot, the sheet metal edges of the "hole" were coamed with leather so that they wouldn't be badly hurt if the flights got a little rough, and a small windscreen -- a couple of inches high as Don recalls -- was installed to deflect the airflow. That was the data recording "office" in the airplane, and there Don and the others recorded data in flight. We suspect that Don is one of the very few people in the world who has logged genuine open cockpit jet time.

Don related a few interesting anecdotes from his open-cockpit time in the P-59. On one occasion, the pilot was in a hurry to get airborne and told Don to come along and never mind bringing his oxygen mask since they wouldn't be climbing above 10,000 feet. Aloft, the pilot became absorbed with the tests and before long he announced to Don over the intercom that he was climbing to 15000 feet. Now it so happened that Don's throat mike had failed and he couldn't talk back to the pilot, but Don figured that shouldn't worry him excessively. After all, he thought, Mt. Whitney is almost 14,500 feet high and ordinary people regularly hiked to its summit. He had only to relax and breathe easily. That approach worked until the pilot announced that he was going to climb to 20,000 feet, at which point Don began to be slightly anxious: the air was already pretty thin at 20,000 feet, and what if the pilot decided to climb to 25,000 feet next? Don considered unbuckling his safety-belt and turning around in his seat to face the pilot. Perhaps he could make some signal which would remind the pilot of his predicament. He finally decided against that idea for two reasons. First, what if the pilot rolled the airplane while he was unbuckled? Second, at that time the P-59 had an endurance of only about half an hour, and since Don had timed the flight as part of his data recording procedures he knew that most of that half hour had already elapsed. He decided to wait it out. Sure enough, within a couple of minutes one engine flamed out, followed shortly by the other engine. They glided back for an uneventful dead-stick landing on the dry lake bed.

After World War II, America's leading aces were invited to Muroc to fly and evaluate the P-59. On one of those evaluation flights Don relates that he occupied the open cockpit cut into the nose of the airplane and Don Gentile occupied the conventional pilot's position. They were testing the airplane over a bombing range near the town of Mojave. They finished their tests a little earlier than expected and Don Gentile asked Don Button what he would like to do. Don suggested they might fly by the house of some friends who lived at Willow Springs, not far from Rosamond. And fly by they did, so low that Don thought they would surely take part of his friends' house or fence with them. According to Don, the flyby was equally memorable for his friends.

Here are some excerpts from another story, told by General Chuck Yeager (Reference 2). The Air Force had taken over the X-1 test program in 1947 and Yeager was the X-1 pilot. To get the feel of the airplane (evaluate its handling qualities?), his first three flights were to be glide flights, carrying no fuel. Remember that the X-1 was carried aloft under a B-29, suspended half within and half outside the B-29 fuselage. Entrance to the X-1 cockpit was gained through the side of its fuselage. Listen to General Yeager:

> "On a typical flight, we would load the X-1 under the B-29 and then service it with liquid oxygen and alcohol. Then the crew of the B-29 would get aboard the aircraft and I, as the X-1 pilot, would always take off in a position just aft of the pilot and co-pilot in the B-29 cockpit. The reason for this was the fact that the X-1, fully loaded, had a stalling speed of around 240 miles per hour and the climbing speed of the B-29 was roughly 180 miles per hour so, if you had an inadvertant shackle release and were in the cockpit of the X-1 at an altitude somewhere below 10,000 feet, you would end up in an inadvertent spin with no time to recover and no way of getting out of the cockpit.

"During the climbout, as we went through 10,000 feet, two sergeants - there were about six - got you by each arm and drug you back into the bomb bay. At this point you put on a seat-type parachute and got on a ladder that was mounted on the right side of the bomb bay. Usually, Jack Ridley would hang on to the cable and let you down on this ladder, which was on two guides which would slide you down into the slipstream opposite the door of the X-1 - which was on the right side. Once you got in, feet first, you would squirm around into position and then put on your shoulder straps and safety belt. Next, they would lower the door down on the guides and hold it against the side of the X-1. You would lock it from the inside using roller locks similar to the type used on a bank vault. At this point you put on your helmet. As Bob Hoover will remember, in the early days and even during the first flights of the X-1, all we had in the way of protective gear was a leather World War II helmet, with the top of a football helmet cut off and snapped onto the top for protection against banging your head on the top of the cockpit, a plain wool flying suit, and a leather A-2 jacket the same as we wore during World War II....

"We began the first three flights on the X-1 and, since I didn't have any experience in that type of aircraft, the first three flights were glide flights without any fuel aboard. **I really didn't learn a lot because immediately after the drop Bob Hoover would jump me in his P-80 and we'd end up in a Lufberry and then roll out on the deck and land.** [Emphasis added.]

The first powered flight that I made in the aircraft, we dropped at 20,000 feet. There were four switches across the instrument panel that were used to ignite the four chambers. I could start and stop any chamber as many times as I wanted - or until I ran out of fuel....

On the first flight, I dropped, ignited one chamber, lit off another chamber, shut that one off, lit another chamber, shut that off, and then, with one chamber running, **pulled up and did a roll** [emphasis added] and found out - my first experience - that you can't, of course, operate with liquid oxygen under zero-g conditions because you get cavitation. The rocket flamed out and as I rolled level again, the sensors took over and re-ignited the igniter and the chamber fired off again. That was my first lesson.

"After checking out the four chambers, I came down in a glide across Muroc, along the old runway, pulled up and lit off all four chambers. We had been told rather seriously by General Boyd ... not to expose ourselves to dangerous situations with the X-1 because safety was a primary factor in the whole program and a lot depended on this airplane. In fact, our whole research and development program in aerodynamics really hinged on our getting the X-1 up above the speed of sound. Well, when I lit off all four chambers, I kept the nose coming up so it looked like the ME-163 coming off the deck. As it got up to about 0.78 - 0.79 - 0.80 Mach, in order to keep the positive g on the airplane, I pulled the nose on through and as the nose started down, the Mach number kept going up, obviously. As I went through 0.82 Mach number - on up to almost 0.84 - I racked off the switches and rolled out. I jettisoned the remainder of the fuel and came down and landed. Jack Ridley and I spent three nights writing a letter to General Boyd in answer to his, "reply by endorsement as to why you exceeded 0.82 Mach on your first flight." that was the most difficult test report I ever had to write because there was no excuse in the world. But, I did feel at home in the aircraft and really everything worked beautifully.

"So the program started...."

The relaxed and unfettered atmosphere which pervades General Yeager's recollection of his early X-1 flights may or may not have been completely factual. We don't know. But can you imagine similar "testing" in today's programs? What if during the space shuttle approach and landing tests the pilots had lifted free of the 747 carrier airplane and **done a roll!** Or had been jumped by a T-38 chase airplane and ended up in a "dogfight"? We suppose it is the sharp contrast between yesterday and today that makes yesterday so intriguing.

EVOLVING STRUCTURE OF HANDLING QUALITIES TESTING AT THE AIR FORCE FLIGHT TEST CENTER

We don't know whether or not the preceding anecdotes are representative of all early handling qualities testing at Wright Field's outpost in the southern California desert. It does appear though that soon after Edwards became the Air Force Flight Test Center, handling qualities testing began to take on a more structured form. We should point out first that handling qualities testing, then as now, was not an isolated, disconnected set of tests. Preliminary handling qualities evaluations were conducted very early in the life of a new airplane, or of a new variant of the airplane. "Complete" handling

qualities and stability and control tests were conducted later, after the Air Force had taken over testing. And of course handling qualities feedback occurred during the "operational" portion of the test program. In those days the testing was formally broken down into "Phases". Later it was called "Category" testing. Now it is called "Development, Test, and Evaluation" and "Operational Test Evaluation". Whatever the testing process is called, and however it is organized on paper, it is historically evident that handling qualities testing necessarily (and beneficially) spans the process.

A survey of early AFFTC test reports reveals a fairly standard approach to formal handling qualities testing. If you were to look at the tables of contents from the early reports you would discover that in nearly every case the following topics would be presented for discussion:

1. Cockpit Evaluation
2. Taxiing and Ground Handling
3. Take-off and Initial Climb
4. Dynamic Longitudinal Stability
5. Trim Changes
6. Maneuvering Flight
7. Dynamic Directional Stability
8. Static Directional Stability
9. Lateral Control
10. Inertial Coupling
11. Stalls and Low Speed Handling Characteristics
12. Approach and Landing
13. Control Friction

Without going into great detail, it is worth noting that items 1, 2, 3, 11, and 12 were mission oriented handling qualities evaluations, in part qualitative (pilot evaluation) and in part quantitative. The qualitative part generally had to do with instrument presentation and location, location and function of switches, problems in taxiing and take-off (e.g.,inadequate nose-wheel steering or braking), problems in maintaining climb schedules, etc. The quantitative part of these items had to do with such measurements as nose-wheel steering break-out forces and hysteresis, rudder and elevator control power during take-off, etc.

The other items were classical up-and-away handling qualities evaluation items. Typically, open-loop, or nearly open-loop tests were used to gather the data. For example, longitudinal static stability data were obtained by flying constant altitude accelerations and decelerations using power to control speed and by flying constant power accelerations and decelerations using stabilizer deflection to control speed. These data were plotted in the familiar form of stick force versus Mach number, stabilator position versus Mach number, etc. Maneuvering flight data were obtained during wind-up turns. These data were presented as curves of stick force versus "g" and stabilator position versus "g". Longitudinal dynamic stability data were obtained using stabilator pulses. The free response of the airplane was measured and used to estimate the short period frequency and damping ratio. Typically these static and dynamic parameter values were compared with the Level 1, 2, and 3 boundaries of MIL-F-8785 (Military Flying Qualities Specification) to determine whether the airplane handling qualities were satisfactory or unsatisfactory.

We do not mean to play down the value of these open-loop handling qualities tests. In fact they were explicity static and dynamic stability tests. They were handling qualities tests only implicitly, and this was widely recognized. However, considerable insight into the handling qualities of the aircraft could be gleaned from these open-loop stability parameters.

Neither do we mean to imply that no pilot-in-the-loop testing was done. In our survey of early Air Force Flight Test Center reports we discovered that air-to-air and air-to-ground tracking evaluations were occasionally reported. In fact, one gets the impression that a fair amount of pilot-in-the-loop handling qualities evaluation was done using air-to-air, air-to-ground, formation, and air-refueling tasks. However, it appears that these evaluations were largely informal, and the results (purely qualitative) appeared in the reports only implicitly. As an interesting aside, some of the earlier color of flight testing during Muroc days seems to have survived in these informal handling qualities tests. Two anecdotes illustrate the more colorful side of these informal, and certainly unofficial handling qualities tests.

There is the story that at one time it was all the rage among Edwards test pilots to make very low, high speed passes down local desert roads. If there were cars on the road, so much the better. It is said that these "tests" ended when the center commander (unbeknownst to the pilot) was run off the road. We cannot vouchsafe the authenticity of that story, but we have heard it often.

Another story concerns an unofficial evaluation of air-to-ground tracking handl qualities. One of the test pilots was making 45 degree dive bombing passes, using local farm house for his "target", when his drop tanks unexpectedly jettisoned. Fortunately, the tanks missed the "target". It was not recorded whether the reason for missing was poor handling qualities or poor drop-tank ballistics.

"War stories" of informal testing aside, here is the important thing to be remembered about handling qualities testing at Edwards from the early 1950's through the early 1970's: during that time, handling qualities tests were almost exclusively open-loop tests. Judgements of the suitability of the aircraft's handling qualities relied heavily on these open-loop test results. *In general, there were no* satisfactory test methods for formally evaluating pilot-in-the-loop handling qualities in a mission oriented setting.

This pattern of fundamentally open-loop handling qualities testing persisted until the early 1970's, when we began to develop the pilot-in-the-loop , mission oriented techniques which have since been named Handling Qualities During Tracking (HQDT) and SIFT .

SYSTEM IDENTIFICATION FROM TRACKING

SIFT test techniques are pilot-in-the-loop, mission oriented test techniques for obtaining and evaluating aircraft handling qualities data. Qualitative pilot comments and quantitative system analysis results are obtained from the same pilot-in-the-loop test maneuvers. SIFT techniques have made it possible to uniquely link qualitative and quantitative test results for the first time.

In Figure 1 we have attempted to give you an over-all perspective of SIFT test and analysis techniques. SIFT is like a broad umbrella: it includes somewhat specialized test maneuvers and piloting techniques for obtaining handling qualities test data; and it includes frequency domain data analysis techniques. Subsequent data analysis includes heavy use of control system and handling qualties criteria.

In the following paragraphs we will briefly cover the development of SIFT techniques, some of the most important features of these techniques (for example, why do the SIFT tracking maneuvers work when earlier, similar attempts did not?), and examples of SIFT test results.

More complete documentation of SIFT techniques is available in References 3 and 4.

DEVELOPMENT OF SIFT TEST MANEUVERS: HQDT

Development of the SIFT test maneuvers and piloting techniques began in 1971. The idea of using air-to-air tracking maneuvers for handling qualities evaluation was adapted by the senior author from similar work which had recently been done by Tom Sisk at the NASA Flight Research Center (Reference 5). (The idea of using tracking as a test tool for evaluating handling qualities was not a new one; for example, see references 6 and 7.)

We used a variable stability F-4C for our development progam. With this airplane we could set up a wide range of handling qualities characteristics, ranging from Level 3 (bare airframe F-4 without any dampers or augmentation: pilot ratings of 7 and 8 on the Cooper-Harper scale) to Level 1 (excellent handling qualities: pilot ratings of 2 and 3). We conducted all of our tests using three handling qualities configurations which corresponded to Level 1, Level 2, and Level 3 according to the pilots' ratings. The configuration could be changed by the test engineer in the rear cockpit, so the pilots were never aware of which configuration they were flying. Two test pilots participated in the development program: Lt. Colonel (now Colonel) Richard E. Lawyer and Major (now Colonel) Cecil W. Powell.

We began our development using five principal test maneuvers:

1. Air-to-air tracking of a target airplane at constant load factor and constant Mach number, with an attempt to maintain constant altitude

2. Air-to-air tracking of a target airplane during a slow wind-up turn at constant Mach number, with an attempt to maintain constant altitude

3. Air-to-air tracking of a target airplane during a maximum roll-rate turn reversal at constant load factor and constant Mach number. The reversals were usually sandwiched between constant load factor tracking maneuvers.

4. Air-to-ground tracking during standard 45 degree and 30 degree dive bombing runs

5. Air-to-ground tracking during standard 15 degree strafing runs

A noncomputing gunsight with a fixed pipper depression was used at all times. This was because we wanted to evaluate the airplane handling qualities rather than the gunsight dynamics.

We measured what we believed would provide us some indication of pilot workload: the time integral of the absolute value of pitch and roll stick forces and rudder pedal

forces. We expected to correlate these three measures of "workload" with pilot comments and tracking errors.

At first, we allowed the test pilots to track the target using any technique they wished. We quickly discovered that this approach did not work. Our "workload" measurements did not correlate with either the pilots' comments or with the tracking errors. Further, the tracking errors did not correlate with the pilot comments. Analysis of those early test results showed that the only reliable data were the pilot comments. First-hand observation from the rear cockpit together with pilot comments soon made it clear why those early results were so confusing. The pilots were working very hard at mental compensation. Because their compensation was mental it did not show up in our physical "workload" measurements. Because their compensation was quite often successful, we could not correlate the tracking error results. And because the pilots were at least roughly aware of the scope and degree of the compensation required, their ratings usually reflected the actual handling qualities of the airplane.

Early on then, we found ourselves faced with the same problems which had plagued earlier investigators. How to construct the tests so that all of the airplane dynamics (within the handling qualities frequency range) would be consistently and repeatably revealed, both to the pilots and to the engineers? Our solution was to have the pilot remove his feet from the rudder pedals and force him to track a precision aim point on the target as aggressively as possible.

By "aggressive" tracking, we mean that the pilot was required to correct even the smallest pipper errors as quickly and as positively as possible. The pipper was never permitted to "float".

Flying without using the rudder pedals removed a major avenue of compensation from the pilot. Aggressively tracking a precision aim point on the target effectively raised the pilot's gain and also reduced his ability to compensate. We cannot over-emphasize the importance of the aggressive piloting technique. It is absolutely fundamental to a good qualitative pilot evaluation and, as we shall see later, to a good system identification of the airplane. A graphic illustration of why the aggressive tracking technique was so important is presented in Figure 2. Figure 2 is a power spectral density (PSD) plot of pilot input (pitch stick force) versus frequency. Two curves are presented. One is the PSD of pilot input when the pilot was allowed to track using any technique he desired. Usually, this involved "floating" the pipper in the vicinity of the target so as not to excite the undesirable short period dynamics of the airplane. When left to his own devices, the pilot's frequency content (that is, the energy which he put into the controller) diminished dramatically at the short period frequency and higher frequencies. Clearly, by "floating" the pipper the pilot was acting as a low pass filter with his cut-off at the short period frequency. Hence he did not excite the undesirable short period dynamics and his tracking errors were relatively small. The second curve is the PSD of pilot input when he was required to track aggressively. The frequency content is noticeably higher at the short period and higher frequencies because the pilot was exciting the short period and higher frequency dynamics. As a result, his tracking errors were larger.

Our experience was that the pilot's comments were more useful when the aggressive tracking technique was used. Our experience suggested that, if left to his own devices, the pilot could report that compensation was required, but he could not necessarily describe what he was compensating for or how he was compensating or what the effect of his compensation was. On the other hand, when the pilots' used the aggressive tracking technique they were able to provide surprisingly detailed descriptions of the airplane response to their inputs. Flying without using the rudder pedals also improved the quality of the pilots' comments, especially when lateral-directional handling qualities deficiencies were present. We found these descriptions to be much more useful in evaluating handling qualities from both a piloting and an engineering viewpoint.

After adopting these techniques we discovered that both pilots' comments and Cooper-Harper ratings became more consistent. A plot comparing the ratings given by the two pilots is presented in Figure 3. We found that the pilots were able to detect surprisingly small changes in the airplane handling qualities and that they were able to quickly identify and describe handling qualities deficiencies. Wide use of these techniques at AFFTC has confirmed these results over and over again.

Each of the maneuvers listed above ,when coupled with the special piloting techniques just discussed, proved to be an excellent test maneuver for evaluating handling qualities. We tried other maneuvers as well, including formation flying, air-refueling, and approach and landing. Our experience was that air-to-air tracking was a "global" test maneuver for evaluating handling qualities, whereas the others tended to be "subsets". In other words, air-to-air tracking was the more demanding maneuver, the more rigorous discriminator of handling qualities characteristics. If handling qualities were optimized for air-to-air tracking, they turned out to be optimized for all other tasks as well. The converse was not true. This suggests the possibility of using air-to-air HQDT techniques as a test maneuver for evaluating the handling qualities of any configuration - even the approach and landing configuration.

Rogers Smith of Calspan has coined the phrase "handling qualities cliff" to describe the problem of handling qualities which are poor but manageable until the pilot gets

backed into a corner (and gets his gain up), at which time "poor but manageable" turns into "disastrous". HQDT has proven to be an excellent tool for ferreting out those "cliffs" during the flight test programs.

For example, during a recent flight test program, a longitudinal pilot-induced-oscillation (PIO) was unexpectedly encountered during a performance test maneuver. Low short period damping indicated the possibility of PIO tendencies, but actual PIOs had never occurred. Indeed, repeated attempts to replicate the PIO using the same and similar test maneuvers were mostly unsuccessful. When SIFT test techniques (air-to-air HQDT) were tried PIOs occurred every time, without exception. In this case, the handling qualities "cliff" had always been present, but it took a set of unintentional, aggravated circumstances to expose it. If SIFT techniques had been used in the first place, the "cliff" would have been exposed immediately.

The special piloting techniques of aggressive tracking and not using the rudder pedals yielded a very good tool for making comprehensive and repeatable qualitative pilot evaluations of handling qualities. However, we had not solved the problem of getting a good quantitative evaluation. We have already said that aggressive tracking without using the rudder pedals caused larger tracking errors. Neither were the errors necessarily repeatable from one maneuver to the next. Consequently, such statistical measures as root-mean-square (RMS) error, cumulative error distribution, time on target, etc. had to be treated very carefully. In general, these statistical measures proved to be unreliable. Figure 4 illustrates why you must be very careful with pipper error as a measure of handling qualities. The best tracking case had the poorest pilot rating and the best rating was associated with tracking results which were not much different from those that got the poorer ratings. Interestingly, the two cases which were both rated 4 were flown in the same airplane, on the same flight, at the same test conditions, by the same pilot, one right after the other. But the tracking errors are distinctly different!

The evidence has been unremittingly in favor of believing the evaluation pilot's ratings and comments. That is why we stressed the value of a running commentary during the test maneuvers and a comprehensive debriefing immmediately following the flight. It proved equally important that the pilots arrive at their Cooper-Harper ratings according to the published approach (Reference 8).

In studying the time histories of pipper motion we noticed an apparent correlation between the dominant frequency content in the piper motion and the pilot ratings. This led to the senior author's suggestion that a frequency domain analysis might yield more reliable quantitative results.

DEVELOPMENT OF SIFT ANALYSIS TECHNIQUES: THE FREQUENCY RESPONSE ANALYSIS COMPUTER PROGRAM

The Frequency Response Analysis computer program, abbreviated FRA, resulted from our effort to provide a quantitative foundation for evaluating handling qualities test results. We found FRA to be an excellent tool for identifying the linear dynamics of the airplane and its flight control system. We also found FRA, together with the various frequency domain handling qualities criteria which are currently available, to be an excellent tool for evaluating handling qualities. The various handling qualities criteria include the equivalent systems criteria, the bandwidth criteria, the Neal-Smith criteria, and the Ralph Smith criteria. (The literature on these criteria is fairly extensive, so we will not discuss them here. See, for example, References 9 through 15.)

FRA is an unusually flexible computer program which can easily be made to do anything the user wishes to do, within quite broad limits. We have used it extensively to identify multiple-input frequency response transfer functions, power spectral density functions, and various coherence functions. Any "system" for which the user has one or more inputs and a response may be identified. For handling qualities testing, the "system" is usually the airframe, or the flight control system (or smaller parts of it), or the combination airframe plus flight control system.

The idea of measuring the frequency response of the airplane is not a new one, either at Edwards or elsewhere. Some of the earlier frequency response work done at Edwards is documented in References 16 through 24. It is worth noting three characteristics of this earlier work: the data were obtained using open-loop test techniques (discreet sinusoidal inputs or pulse inputs); the reduction of the data was extraordinarily time consuming; and the reliability of the results was not computed. Of course, each of these characteristics is directly traceable to the relatively primitive computer capability which was available then. Today's computers permit the use of far more powerful flight test tools. With FRA we can quickly and conveniently use "random" time history data to identify transfer functions and compute an estimate of the reliability of these results.

The constant load factor, constant Mach number, and nominally constant altitude air-to-air SIFT test maneuver proved especially well suited to linear frequency response analysis. The fixed test conditions served to minimize the impact of such nonlinear elements as: stability derivatives which changed with Mach number, angle of attack, and sideslip; changing dynamic pressure; and flight control gains which were scheduled according to Mach number, angle of attack, etc. The aggressive, precision tracking

techniques used by the pilot maximized the pilot's gain, which assured that the airplane dynamics were adequately excited across the handling qualities frequency spectrum. This ensured a good identification of the system.

One of the things which made SIFT techniques so useful was that the data used in FRA was obtained during the same pilot-in-the-loop test maneuvers that produced qualitiative pilot comments and ratings. Another was that the transfer function identification computed by FRA was entirely independent of any preconceived model of the system. These two characteristics of SIFT analysis techniques were especially valuable, making it possible to uniquely link the system identification to the pilot evaluation. For example, in a recent flight test program the pilots reported a pitch "bobble", or oscillation during HQDT testing. The contractor worked to correct this problem by improving the pitch axis of the flight control system, and indeed an improvement was realized. However, a residual 3 to 5 mil oscillation could not be overcome. HQDT data from several air-to-air tracking maneuvers were analyzed using the FRA program. The pitch rate to stick force transfer function was identified. These results are presented in Figure 5. The amplitude peak at about 3 radians/second is the the short period peak. The presence of the second peak, at about 5 radians/second, was unexpected. To make a long story short, the second peak was caused by lateral-directional cross-coupling into the pitch axis. The 5 radians/second peak was at twice the frequency of the dutch roll mode of the airplane. Each time the airplane side-slipped right or left, the airplane pitched up. When the multiple input capability of FRA was used to "remove" the effects of sideslip, rudder deflection, and rolling tail deflection from the pitch rate response, the transfer functions presented in Figures 6 and 7 were identified. The 5 radians/second peak gradually disappeared as the lateral-directional inputs were effectively "removed".

This case is especially interesting because it demonstrated the value of using the same pilot-in-the-loop test maneuver to get both quantitative and qualitative test results. Evidently, the residual pitch oscillation reported by the pilots was caused by lateral-directional coupling into the pitch axis. Extensive classical testing did not identify the cross-coupling. No amount of work on the pitch flight control system alone would have eliminated those last 3 to 5 mils of oscillation.

Another interesting feature of the SIFT analysis techniques was that the same test maneuvers were used to identify the flight control system (or any part of it) and the airframe dynamics, or the combination of the two. During a recent flight test program, SIFT techniques were used to identify the transfer function of normal acceleration to stick force. The identification was not a particularly good one, so we began looking at the individual pieces which made up the whole. We discovered a nonlinearity in the pitch stick force-feel system (presumably hysteresis). When we tried stick position as the transfer function input instead of stick force, we got excellent identifications.

Here is one more example which illustrates the value of the unique linking of quantitative and qualitative results made possible with SIFT techniques. An aileron-to-rudder-interconnect (ARI) evaluation was done using both traditional open-loop techniques and SIFT techniques. The gain of the ARI was changed with the expectation that it would improve lateral-directional handling qualities. The open-loop full-stick-deflection roll tests showed some areas of improvement and some areas of less desirable characteristics. During the SIFT tests, the pilot reported no discernable difference between the two ARI gains. SIFT data analysis showed the ARI transfer function to be highly nonlinear. In this case, it was known that substantial hysteresis existed in the yaw axis. It was apparent from the SIFT tests and analysis that the pilots were almost never getting outside the hysteresis band during precision pilot-in-the-loop tracking tasks. Consequently, they would not have seen any difference between the two ARI gains, and that is exactly what they reported.

We have made attempts to identify the pilot transfer function using SIFT techniques. Those attempts were unsuccessful, evidently because of the way the test maneuvers were structured. As described above, in the SIFT test maneuver the target is essentially passive (non-maneuvering) relative to the tracking aircraft. Hence it is only the tracking pilot's remnant, or linearly uncorrelated input, which creates the continuing pipper error. Henry Jex, Bob Heffley, and Roger Hoh, all of Systems Technology Incorporated, have suggested that the target aircraft be allowed to maneuver in a controlled way, providing an external source of pipper error. We have simulated this scenario on a digital computer and analyzed the results with the FRA program. The results were encouraging. The AFTI-16 flying qualities engineers propose to test this procedure during their flight test program.

CONCLUSION

Handling qualities testing at Edwards has changed quite a lot over the years. We have moved away from what is perceived to be the wilder, less structured testing when the data system was a soldier in an open cockpit cut into an early jet, and when Chuck Yeager was flying the X-1. We saw the development during the 1950's of a pattern of essentially open-loop testing which persisted through the 1960's and is in fact still very much in evidence today. Most recently, beginning in the early 1970's we saw the development and gradual use of the SIFT mission oriented pilot-in-the-loop test and analysis techniques.

These techniques have proven to be very helpful tools for pilot-in-the-loop handling qualities test and evaluation, both quantitatively and qualitatively. They have provided a method for early, quick, and complete identification of handling qualities deficiencies. In practice, SIFT test techniques (also called HQDT) have also been an excellent tool for optimizing flight control systems. SIFT analysis techniques have provided a powerful method for identifying the actual airplane dynamics in terms of linear frequency response. Most importantly, SIFT has forged a unique link between the qualitative pilot comments and the quantitative analysis results. We have briefly discussed several examples which illustrate this unique link. The spectrum of possible applications of SIFT techniques is large, extending well beyond the limited scope of handling qualities testing.

REFERENCES

1. Wolfe, Tom, The Right Stuff, New York, Bantam Books, 1980.

2. Yeager, Charles E., Brigadier General, USAF (Ret), "Operation of the XS-1 Airplane", The Society of Experimental Test Pilots Nineteenth Symposium Proceedings, Volume 12, Number 4, 1975.

3. Twisdale, Thomas R. and Franklin, David L., Captain, USAF, Air Force Flight Test Center, Edwards Air Force Base, California, 93523, Tracking Test Techniques for Handling Qualities Evaluation, May 1975, AFFTC-TD-75-1.

4. Twisdale, Thomas R. and Ashurst, Tice A., Air Force Flight Test Center, Edwards Air Force Base, California, 93523, System Identification From Tracking (SIFT), A New Technique For Handling Qualities Test and Evaluation (Initial Report), November 1977, AFFTC-TR-77-27.

5. Sisk, Thomas R., Kier, David A., and Carr, Peter C., Factors Affecting Tracking Precision of Fighter Aircraft, NASA-TM-2248, November 1970.

6. Blackburn, A. W., "Flight Testing Stability Augmentation Devices for High Performance Fighters", Society of Experimental Test Pilots Quarterly Review, Volume 1, Number 1, Summer 1957.

7. Bull, Gifford and Kidd, Edwin A., Air-to-Air Tracking With Linear and Non-Linear Yaw Damping, WADC Technical Report 55-223, June 1955.

8. Cooper, George E. and Harper, Robert P., Jr., The Use of Pilot Rating in the Evaluation of Aircraft Handling Qualities, NASA TN D-5153.

9. Hodgkinson, John, Equivalent Systems Criteria for Handling Qualities of Military Aircraft, AGARD Flight Mechanics Panel Meeting on Criteria for Handling Qualities of Military Aircraft, 19-22 April 1982.

10. Hodgkinson, John, and Wood, J. R., Equivalent Systems Criteria for the Flying Qualities MIL Standard, Joint Flying Qualities/Flight Controls Symposium, Dayton, Ohio, 2-5 March 1982.

11. Hoh, Roger H., Mitchell, D. G., and Hodgkinson, John, Bandwidth - a Criterion for Highly Augmented Airplanes, AIAA Paper Number 81-1890, AIAA Atmospheric and Flight Mechanics Symposium, 19-21 August 1981.

12. Neal, Peter T., and Smith, Rogers E., An In-Flight Investigation to Develop Control System Design Criteria for Fighter Airplanes, Vols. I and II, AFFDL-TR-70-74, December 1970.

13. Chalk, Charles R., CALSPAN Recommendations for Supersonic Cruise Aircraft Flying Qualities Design Criteria, NASA-CR-159236, April 1980.

14. Chalk, Charles R., and Weingarten, N., Inflight Investigation of Large Airplane Flying Qualities for Approach and Landing, AFWAL-TR-81-3118.

15. Smith, Ralph H. And Geddes, Norman D., Handling Quality Requirements for Advanced Aircraft Design: Longitudinal Mode, AFFDL-TR-78-154, August 1979.

16. Andrew, G. M. and Polve, James H., Captain, USAF, Trapezoidal Approximation of Transient Flight Data to Obtain Aircraft Frequency Response, Air Force Flight Test Center, FTD-51-24, 1 November 1951.

17. Mullens, Marshall E., Longitudinal Frequency Response Characteristics of a B-25J Aircraft as Obtained by the Transient Method, AFFTC 52-5, 1 March 1952.

18. Andrew, G. M. , The Dynamics of the Automatically Controlled Airplane, AFFTC 52-24, 1 August 1952.

19. Smetana, Frederick O., 1Lt., USAF, An Explanation of the Theory and Applications of

Dynamic Response Flight Testing, AFFTC TN R-17, 15 May 1954.

20. Smetana, Frederick O., 1Lt., USAF, Determination of Frequency Response Characteristics, Including Coupling Tendencies, During Standard AF Phase Tests, With Results on an F-86D at M=.85, AFFTC-TR-54-19, May 1955.

21. Mullens, Marshall E., XB-52 Longitudinal Dynamic Response Characteristics, AFFTC-TR-55-23, June 1955.

22. Smetana, Frederick O., Longitudinal & Directional Frequency Response Characteristics of the F-86D, AFFTC-TN-55-29, December 1955.

23. Starr, Donald W., An Approximate Method of Determining Frequency Response From Non-Periodic Time Functions, AFFTC FTFF-TM-56-19, August 1956.

24. Simmons, Carl and Ballentine, Wilbur A., Dynamic Response Characteristics of the C-130 as Determined During Phase IV Flight Tests, With an Evaluation of the Test and Analysis Procedures, AFFTC-TN-57-18, September 1957.

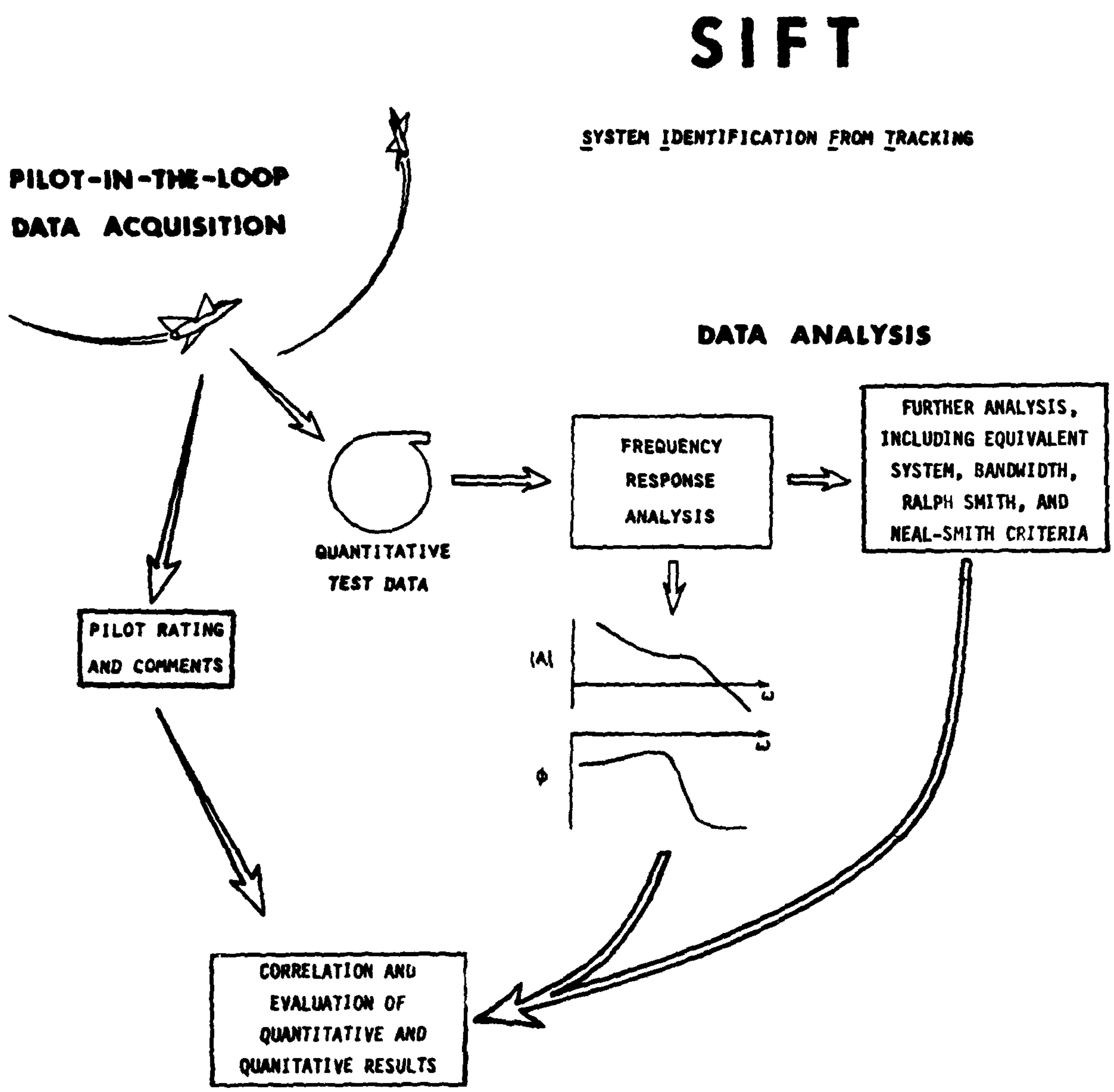


FIGURE 1 SCHEMATIC OVERVIEW OF THE SIFT PILOT-IN-THE-LOOP HANDLING QUALITIES TEST TECHNIQUES

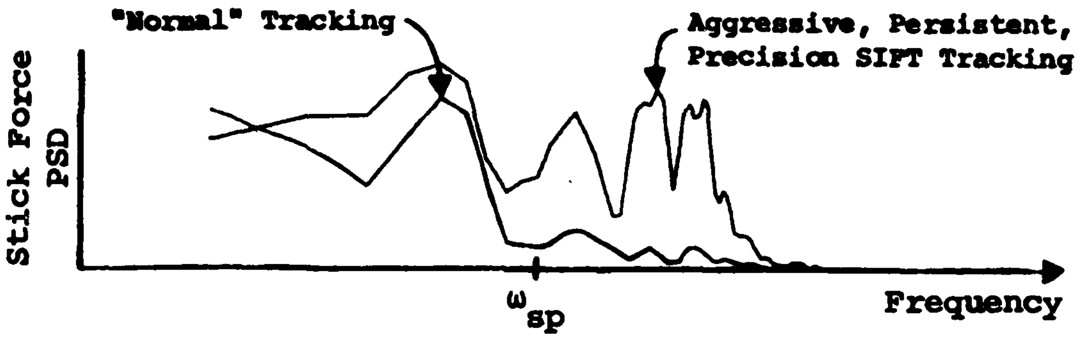


FIGURE 2 COMPARISON OF PILOT INPUT FREQUENCY CONTENT USING AGGRESSIVE SIFT TRACKING AND USING "NORMAL" TRACKING TECHNIQUES

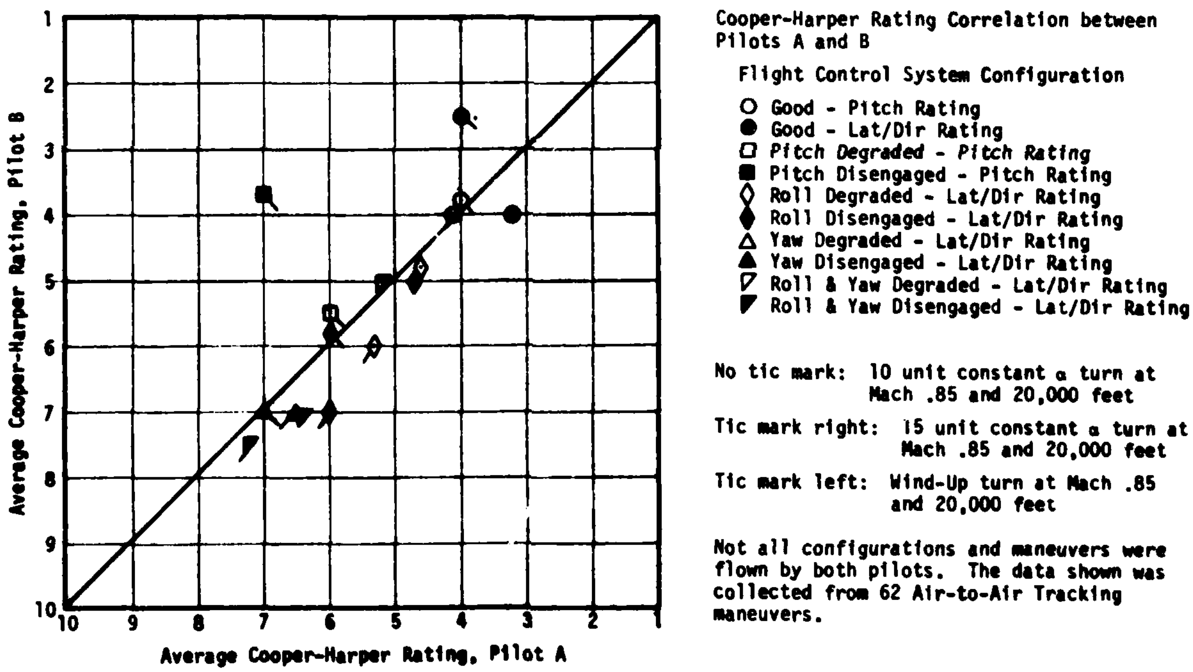


Figure 3. Cooper-Harper Rating Correlation Between Pilots A and B Obtained Using SIFT Techniques

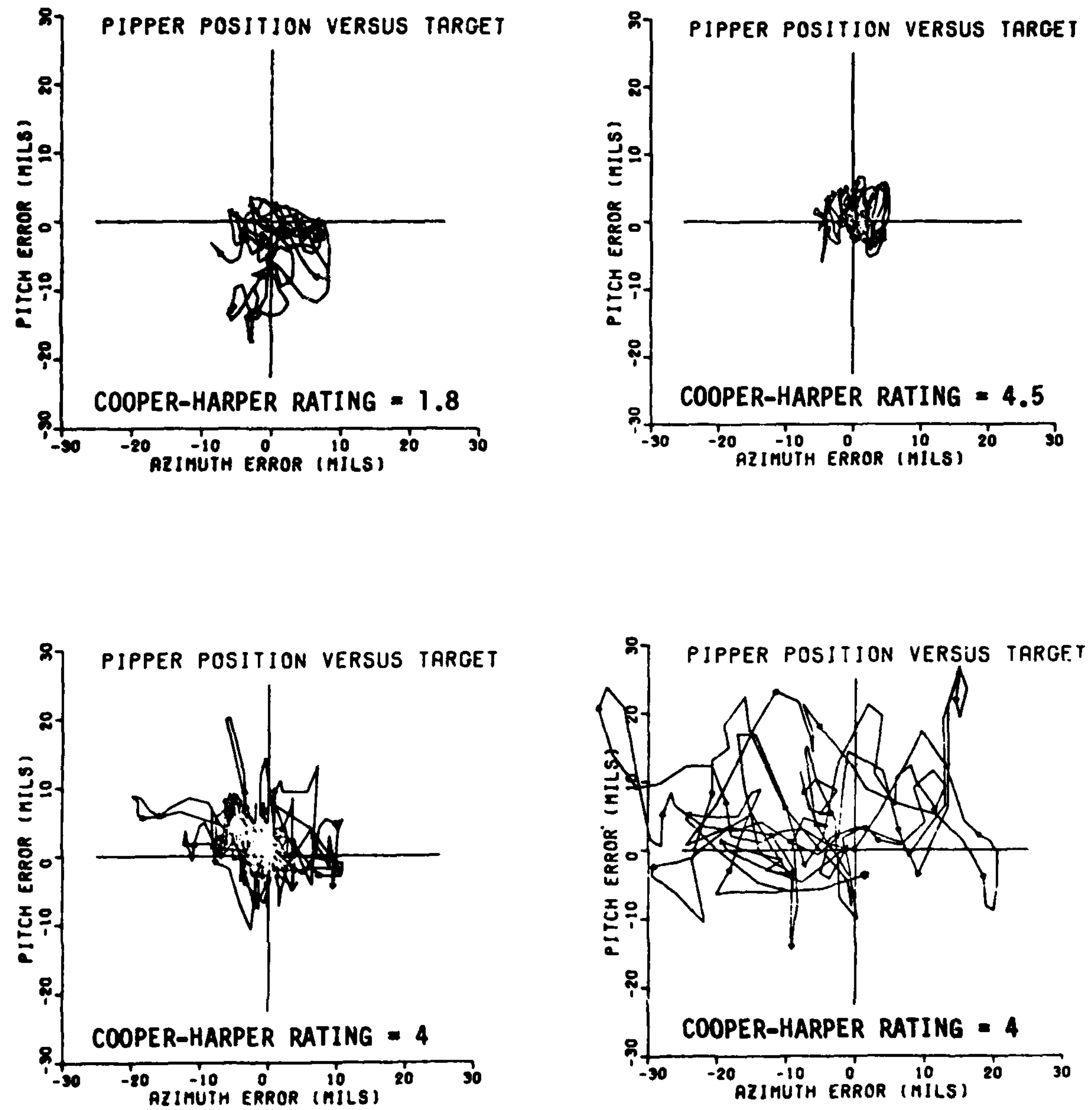


FIGURE 4 EXAMPLES OF TYPICAL SIFT (HQDT) TRACKING RESULTS SHOWING HAZARDS OF RELYING ON TRACKING ERROR AS THE ONLY MEASURE OF HANDLING QUALITIES

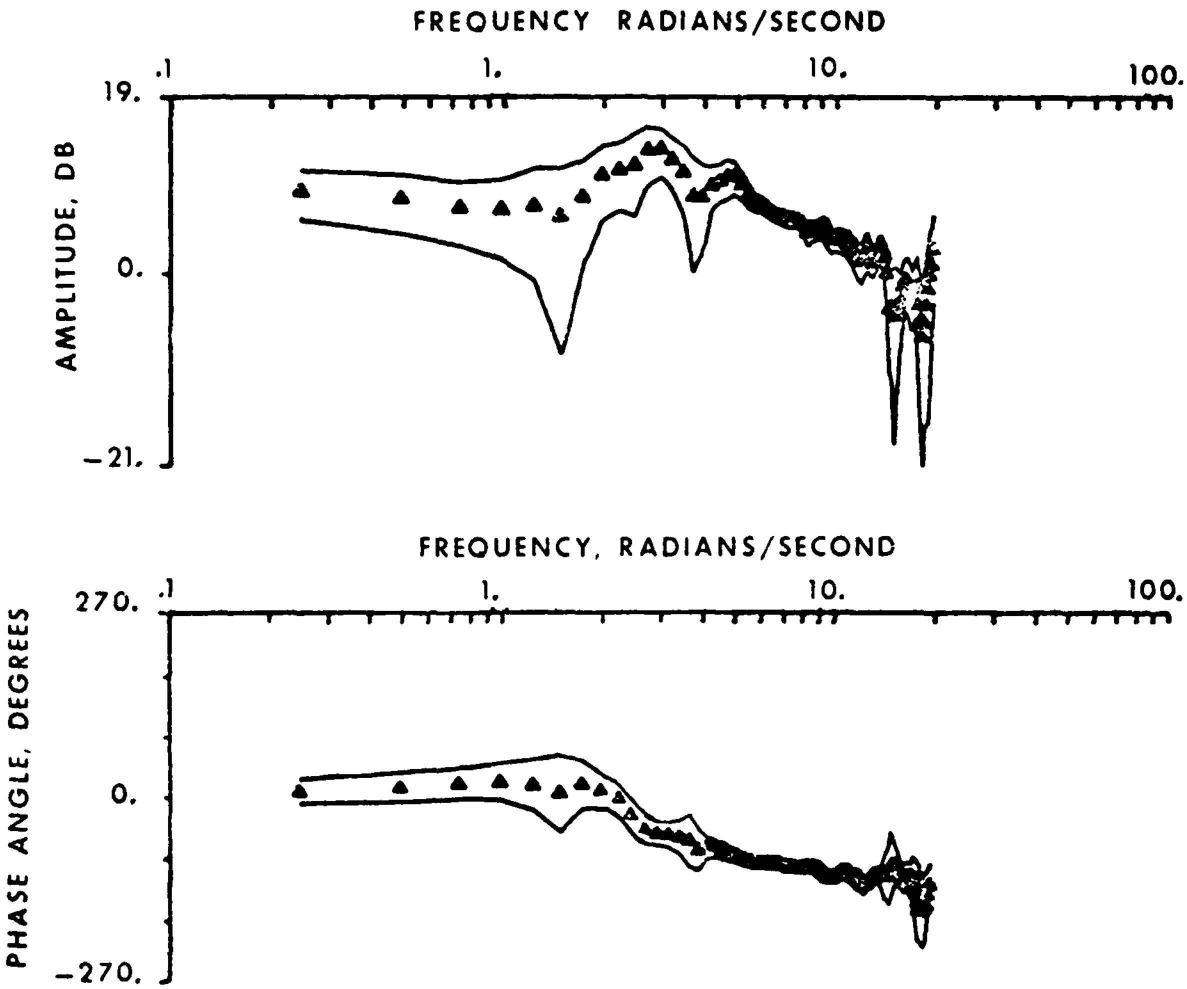


FIGURE 5 TRANSFER FUNCTION OF PITCH RATE TO STABILATOR DEFLECTION, IDENTIFIED USING SIFT TECHNIQUES

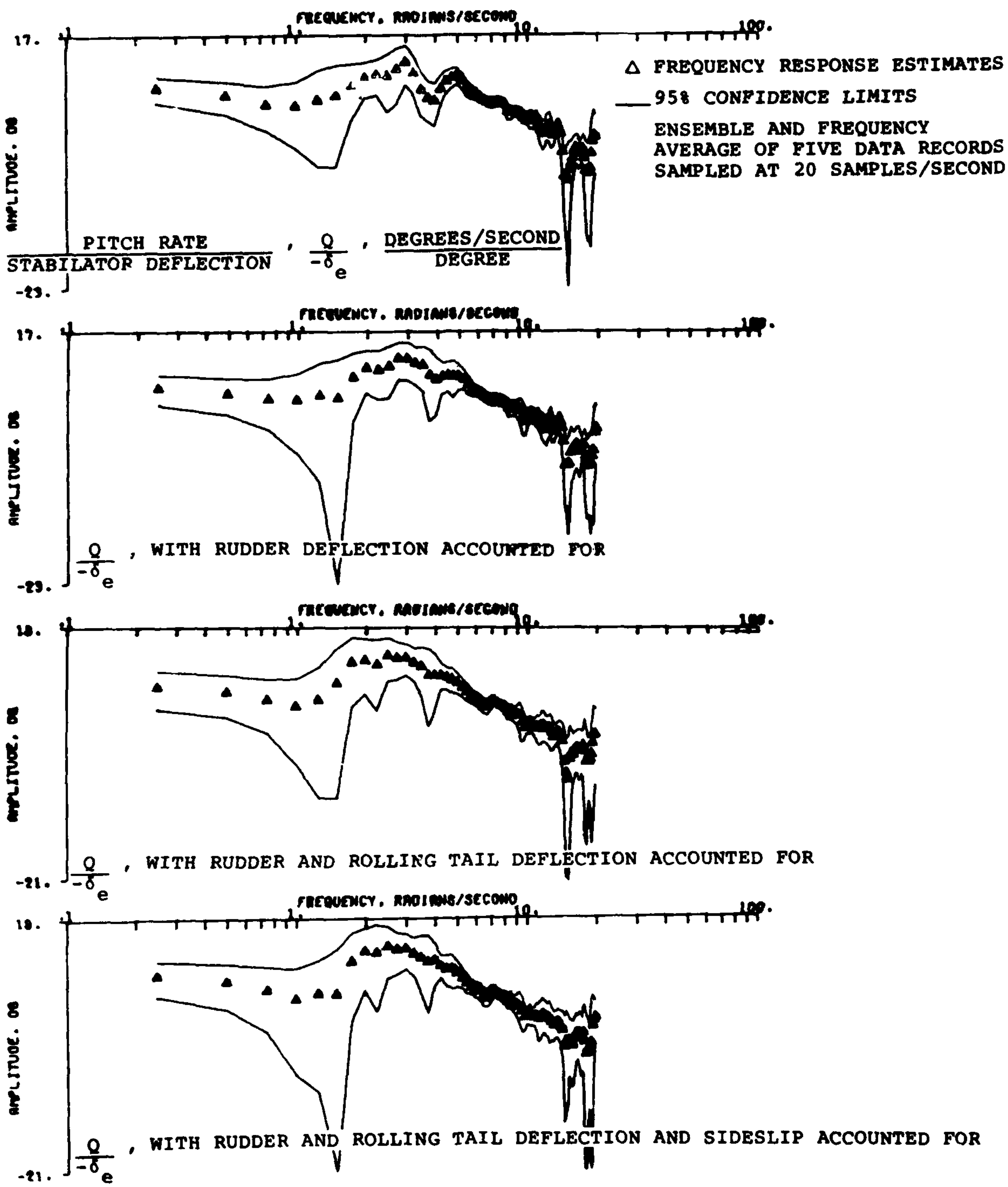


FIGURE 6 BODE AMPLITUDE PLOTS OF PRECISION AIR-TO-AIR TRACKING DATA, DEMONSTRATING SOURCES OF LATERAL-DIRECTIONAL COUPLING INTO THE PITCH AXIS

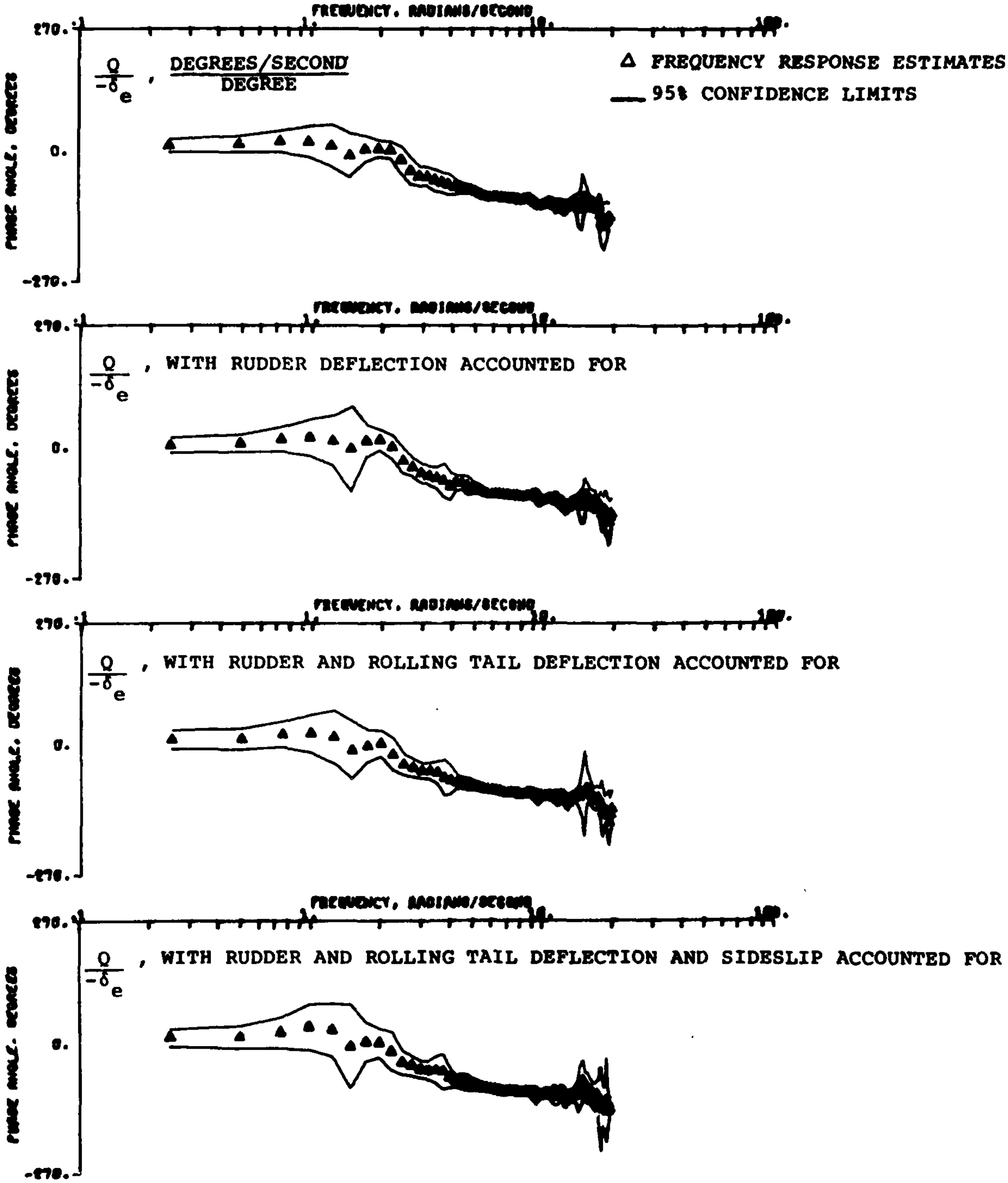


FIGURE 7 PHASE ANGLE PLOTS CORRESPONDING TO AMPLITUDE PLOTS IN FIGURE 6

EXPERIENCE WITH SYSTEM IDENTIFICATION FROM TRACKING (SIFT) FLIGHT-TEST-TECHNIQUES AT THE GERMAN AIR FORCE FLIGHT TEST CENTRE

(Flight test evaluation of the PIO - behaviour of a helicopter with suspended cargo)

by

E. Buchacker
Bundesamt für Wehrtechnik und Beschaffung
AFB LG IV
Flugplatz
D 8072 Manching
West Germany

SUMMARY

The report deals with pilot induced oscillations (PIO) which occured mainly during landing approach of a medium size cargo helicopter with a suspended load. The flight test programme which was set up to gain insight in the problem is briefly described.

Data evaluation showed that a bad combination of eigenfrequencies from a suspended load and the helicopter caused a very poorly damped eigenmode. This mode could be excited by the pilot but was not controllable for a human being because of the frequency ($\sim$ 11 rad/sec) involved.

A good correlation between pilot comments and flight test data evaluation was found.

CONTENTS

1. INTRODUCTION

During use of the CH-53 helicopter to transport externaly suspended cargos, severe oscillations occurred repeatedly and especially during the approach phase. Based upon pilot statements, these oscillations must be classified as "oscillations of the closed loop (ie,pilot/helicopter)". In some cases it was even necessary to drop the cargo in order to restabilize the helicopter.

The objective of the trials was to clarify how these oscillations develop and to give hints on the avoidance or the prevention of PIO.

2. THEORETICAL MODELS WITH REGARD TO A HELICOPTER WITH SUSPENDED CARGO

The following is an attempt to isolate the type of oscillations leading to PIO by means of a few simple models. Two types of cargo oscillations related to the helicopter seem to be important in the connection considered here:

a) pendulum oscillations of the cargo
b) vertical oscillations of the cargo initiated by spring type characteristics of the cargo sling.

In order to be able to still present the relations in a simple and clear manner, the following considerations are based upon the commonly used simplifying assumptions of small angles; concentration of masses in the individual center of gravity; absence of damping; linear system behaviour and massless springs or suspensions; because the pendulum oscillations proved not to be important for the case considered here they are not discussed further.

2.1 VERTICAL OSCILLATIONS OF THE CARGO

Fig. 1 shows the model for estimating vertical oscillations of the system helicopter plus external cargo.

On the CH-53, the cargo hook is not directly connected with the gear box or the rotor; but the load is routed via the fuselage box to the rotor. The fuselage box is comparatively flexible. The model depicted in Fig. 1 approximates the fuselage to a spring and mass system (spring constant c_1 and partial helicopter mass M). The cargo with mass m is attached by means of the cargo sling with the spring constant c_2 to the partial helicopter mass M. This dual mass and dual spring system is carried by the drive unit of the helicopter. Forces which may excite the system to oscillate are the air and mass forces produced by the rotor. These are the periodic forces as a result of the rotor rotation on the one side and the responses to the pilot inputs on the other side.

With the frequencies

$$w_1^2 = \frac{c_1 + c_2}{M} \qquad w_2^2 = \frac{c_2}{m} \qquad \text{and the mass ratio} \qquad \mu = \frac{m}{M}$$

the following is obtained according to /1/ for the two natural frequencies of the system:

$$W_{1,2}^2 = w_1^2\left[\frac{1}{2}\left(1 + \frac{w_2^2}{w_1^2}\right) \mp \sqrt{\frac{1}{4}\left(1 - \frac{w_2^2}{w_1^2}\right)^2 + \mu\frac{w_2^4}{w_1^4}}\right]$$

In Fig. 2 the histories of the natural frequencies ($W_{1,2}$) are presented versus the mass ratio (μ). Approximately typical values for the calculation of the CH-53 with external cargo were assumed to be

$$w_1 = 60\,\frac{\text{rad}}{\text{sec}} \qquad w_H = \frac{w_2}{w_1} = 0.4;\ 0.2;\ 0.1$$

This corresponds to a spring constant ratio of approximately 1:10 for the helicopter and cargo sling. The assumption that only a part of the helicopter fuselage mass is to be considered to be an oscillating mass justifies the extension of the calculation to mass ratios of $\mu = 2$. Results of the flight trials verify that the orders of magnitude used for the calculation were realistic.

The two natural frequencies are close to the frequencies w_1 and w_2 in the range considered here (Fig. 2). The upper natural frequency which is approximately the natural frequency of the fuselage is well beyond the range that can be actively influenced by the pilot (~60 rad/sec).

The lower natural frequency coincides closely with the natural frequency of suspension and cargo (w_2) in the range considered. Depending on the spring constant of the cargo sling and the suspended load, frequencies in the range from 3 to 23 rad/sec may result. In this range, the pilot is still able to respond to the oscillations occurring by control inputs; but at frequencies above 3 to 6 rad/sec, the rate of response

of the man is generally no longer adequate to permit suitable and well adjusted control inputs. As it can be assumed, based upon the design of the cargo slings and the cargo shapes, the damping forces produced in the system are only minor forces, and all conditions for the occurrence of a PIO are given in this case.

Considering the frequencies of the exciting forces that are present on the helicopter independent of the pilot inputs, one will note that the main rotor frequency (approx. 19.4 rad/sec according to /2/) and the difference of the main rotor frequency and the lead-lag frequency (0^{st} lead-lag frequency of approx 5.9 rad/sec according to /2/) are in the range of interest. Thus, "air resonance phenomena" may be aggrevating the problem.

As a whole, it may be concluded from the above considerations that the problems experienced may be caused by the possible vertical oscillations of the helicopter/external cargo system.

3. TEST METHOD

3.1 MEASUREMENT AND ASSESSMENT PROCEDURE

When investigating man/machine systems, as in the present case, it is essential to apply test methods which provide a good qualitative observation of the system behaviour by the participants in the test (the pilots) during the test sequence as well as a technical evaluation and assessment of the data achieved.

3.1.1 QUANTITATIVE EVALUATION

Since a best assessment of a system by the man (pilot) is only possible when performing a task while using the system which requires an exact adherence to given values (such as target tracking tasks), it was obvious that the data, (which was rather random in character); obtained in this case could not be assessed using the classical methods of flight test techniques.

From fixed-wing tests in the USA /3/ and from own results /4/, it was known that the procedures to analyse random events in the frequency range /5/, /6/ could be successfully used for identifying the system behaviour of aircraft. The time histories of the parameters of interest as measured during the flight test were transformed into the frequency range by means of the Fourier analysis. As a first result, power spectral densities were obtained which permitted a statement on the preferred occurrence of certain frequencies to be made.

For example, from the comparison of the power spectrum density of the normal acceleration with the power spectrum of the movement of the collective stick, it can be concluded whether the pilot actively contributed to an oscillation. If he did this can be seen by power peaks at the frequency in question occurring in the power spectrum densities of the collective stick movement.

But in the system input and output power spectrum densities (i.e., "collective stick movement by the pilot (system input)" and "normal acceleration at the pilot seat (system output)") the total information on the system behaviour may also be found. Therefore, the system behaviour may be derived and plotted in the form of Bode or Nichols diagramms from the comparison of the output and input (cross power densities) and the power densities of the input and output. Ref. /7/ recommends the following pilot model as adequate for investigating critical stability conditions:

$$V_p = K_p \cdot e^{-j w t_t}$$

where

V_p = transfer function of the pilot
K_p = gain factor of the pilot
w = angular frequency
t_t = delay time of pilot
j = $\sqrt{-1}$

By adding the pilot model to the derived system transfer functions, it is possible to obtain statements with regard to the stability of the loop closed by the pilot from the Nichols chart.

3.1.2 QUALITATIVE ASSESSMENT

From the fixed wing tests experiences available, an air-to-air tracking manoeuver lent itself as a flight task to be assessed. An Assessment Form replaced an assessment according to Cooper-Harper in the present test series. A List of Questions and the Assessment Form had to be completed by the pilot after the flight or a series of flights with, for instance, varying cargo mass.

3.2 PARAMETERS MEASURED

Fig. 3 shows a summary of the test parameters used for this program and is an extract from the available parameters.

Since the first flight trials have proven that the Automatic Flight Control System (AFCS) itself did not introduce the PIO, the analysis of the behaviour of the autopilot was not continued. The accuracy and quality of the analysis carried through in the frequency range concerned depends strongly on the resolution of the parameters and less on the linearity and the temperature behaviour. Furthermore, the conversion of the analog signals into digital form played a decisive role. Here it should be noted that the proper selection of anti aliasing filters prevents a backfolding of signal portions of higher frequencies in the frequency range that can be covered by digitalisation and, in addition, the relations of the signals to each other is not manipulated due to different phase shift.

The resolution of the signals depends on one hand on the design of the transducer itself and, on the other, on the bit number of the available analog-to-digital converter (ADC).

The ADCs available at E-61 are designed for 12 bits and thus resolve the range of a signal in 4096 steps. Therefore the limit resolution for all parameters 's given by the transducer design.

The Tracking manoeuvres used in the test program utilize the available parameter ranges up to approx. 10 to 20 percent. This results in a resolution from 1 to 7 percent of the range of interest.

Empirically the dynamic range in which the pilot is able to operete is approx. 30 db, i.e. approx. 30:1. This range cannot be fully covered with the available instrumentation system. This limited the achievable quality of the analysis. This can be seen from the diagrams by the partially far deviation of the confidence limits of 95 % from the estimated test points.

3.3 TEST CONDUCTED

Fig. 4 presents a summary of the flights evaluated in this report.

The variables considered during this evaluation were: The cargo mass, the proporties of the sling, and the friction at the collective.

The manoeuvres flown were tracking tasks, including acceleration and deceleration, landing approach and hover. The target was a UH-1D.

4. ANALYSIS OF THE TEST AND MEASURING RESULTS

4.1 ASSESSMENT OF MEASUREMENTS

According to ref. 7 the transfer behaviour, "normal acceleration due to control movement", plays a decisive role for the occurrence of PIO's during target tracking manoeuvres with the main task being to stabilize the attitude of the aircraft in relation to a reference (another aircraft, ground). Therefore this transfer function is especially discussed below.

The Fig. 5 and 6 show the power spectrum densities of the normal acceleration (14) and collective stick movement (04) of a flight without cargo (flight 14) and a flight with heavy cargo (flight 30) for the test phase I (flight in one direction with heading changes of approximately 30° to the right and left hand side) and phase IV (hover).

For both flight phases of flight 14 the power peaks are clearly recognizable in the normal acceleration spectrum density at approx. 20 rad/sec which are caused by the main rotor. Another peak appears at approx. 11 rad/sec in test phase I. Its cause might be the difference frequency of the main rotor frequency and the lead-lag frequency of zeroth order. However, in both cases the main rotor frequency is dominating (larger amplitude).

This situation is changed considerably (Fig. 6) when the helicopter is flown with a suspended cargo. As compared to the amplitude at approx. 11 rad/sec the amplitude of the main rotor frequency decreased considerably. The energy in the system is now increasingly concentrating at the lower frequency. However, this range still stimulates the pilot to take a controlling action where a suitable action would generally exceed his capability.

In addition to the two frequencies already discussed further peaks occur between these two (flight phase IV, Fig. 6) which are probably caused by the vertical automotion of the oscillating system, i.e. helicopter with suspended cargo.

Fig. 7 and 8 show a summary of the frequency peaks in the range from 10 to 20 rad/sec (1.6 to 3.2 Hz) as indicated by the power spectrum densities of the normal acceleration for the various test flights. Since the histories for all test phases are generally similar, only the analyses for test phase I and IV are shown as an example.

The main rotor frequency at 3.0 to 3.1 Hz is clearly recognizable. Below this frequency, a whole series of frequency peaks appear which are distributed so that it is impossible to state unambiguously with the means available here which one of the two natural frequencies can be derived from the difference of the main rotor frequency and lead-lag frequency or the one from the helicopter/external cargo system.
However, a close relation to the frequencies estimated in para 2.1 and those taken from ref. 2 is obvious.

As there is a possibility that energy is transported into the helicopter/external cargo system at the difference frequency "main rotor minus lead-lag", a considerable damping reduction at least will take place.

Fig. 9 to 12 show the transfer functions normal acceleration (14) due to collective stick movement (04) as a Bode diagram for the flight phase I and IV for flight 14 and 30 already used as an example

Disregarding the values above 20 rad/sec, which vary considerably due to inadequate data quality in this frequency range, maximum transfer ratios are found in the range of 8 rad/sec in flight 14 (without cargo).

The order of magnitude of the phase rotations occurring in this case is 90°. The combination of the transfer behaviour found with a simple pilot model, as shown in para 3.1.1, shows that the overall loop "pilot and helicopter" remains stable and the pilot can accomplish his control task without any extraordinary problems. Fig. 13 and 14 contain the Nichols charts prepared for estimating the behaviour of stability.
For a helicopter with external cargo (flight 30, Fig. 11 and 12), the situation is completely different. The maximum transfer factor occurs now at approximately 12 rad/sec and is relatively considerably higher than for the helicopter without cargo. Therefore, the attachment of an external cargo unfavourably effects the helicopter characteristics in two ways:

a) The maximum transfer factor is shifted to higher frequencies.
b) The system damping is reduced.

Both considerably complicate the task of the pilot to control the helicopter. The maximum transfer factor is now in a range in which the pilot is no longer capable of controling the helicopter properly. The Nichols

charts (Fig. 15 and 16) for the flight phases considered also show clearly the reduction of the stability of the closed loop "pilot/helicopter with external cargo".

During the flight test, a PIO occurred which forced the pilot to drop the cargo. Fig. 17 to 19 show the transfer function "normal acceleration due to stick movement" measured in this case. Fig. 20 shows a detail of the associated time histories. From the comparison of the power spectrum densities of normal acceleration (14) and collective stick movement (4) (Fig. 17) as well as from the time histories (Fig. 20), the response of the pilot to the motion which has the effect of maintaining the oscillation can be seen.

From the ratio "maximum transfer factor to transfer factor in the stationary state", the damping of the system can be estimated.

For a second order system, the relations for a damping (ζ) according to ref. /8/ are

$$M = \frac{1}{2\,(1 - \zeta^2)^{0.5}} \qquad \text{and from that} \qquad \zeta = (0.5 - (0.25 - \frac{1}{4M^2})^{0.5})^{0.5}$$

As a stationary value for determinating M, a mean value taken from the initial values of the transfer functions "normal acceleration due to stick movement" was used.

The damping values *) found are summarised in Fig. 21 and 22. While Fig. 21 shows the influence of the test phase, i.e. the influence of the flight condition with the cargo as parameter by means of four examples. The influence of the cargo and the cargo attachment can be seen from Fig. 22. The considerable damping reduction of the system with increasing external cargo is clearly recognisable. (Fig. 22). The one flight with an internal cargo (No 15) does not permit an unambiguous estimation of the influence of internal cargos.

It can also be seen that landing approach and hover phase with large external cargos are critical conditions as far as the damping is concerned (Fig. 21). For flight phase I, the results are partly quite unusual. It seems that this flight phase stimulates the pilot the least to respond by more frequent collective stick movement. This affected the quality of the data.

4.2 ASSESSMENT OF PILOT STATEMENTS

Pilot reports are available on two incidents at the Services and an emergency drop during flight testing at E-61 as a result of vertical oscillations. The essential parts of the reports are briefly repeated here:

> Chase A.
> "While initiating the descent (at a forward speed of 30 to 40 KIAS) severe vertical oscillations connected with abrupt oscillations (approx. 15°) about the longitudinal and transverse axis of the helicopter occurred .....
> In order to bring the helicopter back into a safe flight condition again I increased the power with the result of even increasing vertical oscillations. Therefore I ordered ..... : "Drop". After that the flight condition was immediately normal again."
>
> Chase B.
> "After having passed a power transmission line the descent to approach the landing field was initiated. When changing the collective pitch stick position the helicopter started to oscillate vertically so severely that the flight attitude could no longer be maintained. The amplitude of the oscillations increased rapidly. I tried to balance out the oscillation by an immediate climb, however, without any success. The oscillations became so severe that the helicopter threatened to attain an uncontrollable condition.

---

*) Due to the rather arbitrary selection of the stationary condition the damping values found should not be interpreted to be absolute but rather relative values.

I told my copilot "Drop the cargo". After having dropped the cargo the helicopter immediately returned to a normal attitude."

Case C. External cargo mass 6600 kg

"In addition, during hover at a height of 150 ft inputs of approx. 2 cm in the direction "push" were applied to the collective stick in order to check the oscillating behaviour of the cargo in the vertical direction after the measuring program had been completed. After the 3rd input of unchanged magnitude the helicopter started a no longer controllable "vertical bounce" so that the external cargo had to be dropped in order to correct this condition. From the initiation until the drop a period of approx. 3 seconds was required; all hydraulic warning lights were illuminatod and extinguished after having dropped the cargo. The external cargo was destructed but the helicopter has not been damaged ....."

Vertical oscillation without excitation by the collective stick did not occur (extremely still air). All vertical oscillations produced by collective stick excitation were of small amplitude and could be corrected again by systematically "freezing in" the controls and by releasing the collective trim, respectively, with the exception of the last test in hover during which the helicopter continued the vertical bounces with a constant large amplitude even after trim release.

All three incidents have in common the heavy external cargo mass, the excitation of the oscillation by the transfer of a vertical interference to the system, the feeling of the pilot not to be able to correct the condition by normal or systematic means, respectively, as well as the decay of the oscillation after having dropped the cargo.

According to the results from Fig. 21 and 22, the heavy external cargo must have led to a considerable damping reduction of the helicopter/external cargo system. The attachment of the external cargo produced an additional natural frequency in the system resulting in the susceptibility to air resonance. This can be seen from the comparison in Fig. 5 and 6 as well as from the theoretical considerations given in para 1. This additional natural frequency is beyond the range normally used by the pilot for control purposes. However, an excitation at this natural frequency is possible by somewhat abrupt or "jerky" inputs by means of the collective pitch stick as deliberately used in case C.

In the two other cases it can also be assumed that the descent was made in such an abrupt manner that the oscillation was excited. The systematic behaviour in the cases A and B (initiation of a climb), which can counteract pendulum oscillations of a cargo, instead, causes, the contrary during vertical oscillations. Due to the air resonance mechanism a power increase will also introduce a higher power in the oscillation.

Obviously in none of the cases described above, the collective pitch stick has been released completely. Therefore no statement can be made whether such a procedure would result in the decay of the oscillations. Using the measured results as a basis, this seems to be possible. However, the measured results do not consider the effects of the non-linear behaviour due to the oscillations (such as the loss of hydraulic pressure). Furthermore, it must be doubted that such a behaviour (release of the collective pitch stick) in the given overall situation "landing approach" is practicable.

Since the dropping of the cargo is changing the characteristics of the system entirely, this leads to the correction of the dangerous flight condition according to the test results presented in this report.

From a general point of view, all of the three quoted pilot comments describe the course of such an event in a way that can also be expected from the measured results and the theoretical considerations.

Two pilots of the test centre participated in the second test phase (flight 24 to 31) and completed assessment forms. Fig. 23 summarises the assessments of the pitch, roll and vertical control.

The plot of the assessment values versus the cargo mass shows a rather slight, ununiform tendency to a poorer assessment with increasing cargo mass (pitch and roll motion) for all cases. This also corresponds with the conclusions that can be drawn from the transfer function found for these motions.

Quite unambiguously, the degradation of the vertical control characteristics was assessed by the pilots. The constant difference between the assessment of the two pilots is of course more a personal constant. The unambiguous tendency should be considered here as essential fact.

Summarising, it can be stated that pilot comments and assessment correlate in a quite unambiguous manner with the results found by measurements.
From these results, the system behaviour as measured correctly describes the PIO phenomenon and can therefore also be the basis for a change of the system with the objective to reduce or eliminate the PIO susceptibility.

5. CONCLUSIONS AND RECOMMENDATIONS

The generation of a PIO is generally initiated by inputs in the vertical direction as for instance typical during landing approach (such as collective stick inputs, speed changes and gusts). As the overall system is damped only very slightly in this case and the PIO frequency occurs in a range that can no longer be controlled by the pilot; only dropping the cargo will, especially in the case of heavier external cargos, generally result in the recovery from the dangerous flight condition.

In principle, a better damping of the helicopter/external cargo system or a displacement of the natural frequency generated by the attachment of the cargo towards lower frequencies are conceivable as corrective actions.

A reduction of the PIO susceptibility can also be achieved by making it harder to the pilot to rapidly adjust the collective stick as necessary. This would be possible by increasing the collective stick friction. However, in such a case it should be considered that an increase of the hysteresis width in the actuator, which would result in an increase of friction, has in principle, a destabilizing effect. This corrective action represents an aid for the pilot in freezing in the collective stick position to correct a PIO rather than an improvement of the stability of the pilot/helicopter loop. The upper limit of the adjustment values proposed by the contractor should, therefore, by all means be considered.

Markedly steady flying with slow control inputs especially during the landing approach and landing phase is also suitable to hamper the occurrence of a PIO situation.

6. REFERENCES

/1/ Magnus, K., Schwingungen
(Oscillations), Teubner Studienbücher 2nd edition, Stuttgart 1976

/2/ CCG-Kurs Technik der Hubschrauber
(CCG Course Helicopter Engineering)
Course OF2 of Carl-Cranz-Gesellschaft e.V. of September 1973

/3/ Twisdale, Thomas R., Jones, Gerald L., and Ashurst Jr.;
Tice A, A Mission Oriented Flight Test Technique for Identifying Aircraft and Flight Control System Transfer Function, AGARD-CP 223, April 1977

/4/ E. Buchacker, Flugeigenschaftsuntersuchungen am MRCA TORNADO im Rahmen der amtlichen Flugerprobung, Teil I, Vortrag auf dem DGLR Symposium "Das MRCA-Programm TORNADO", 1979
(Analysis of Flying Qualities of MRCA Tornado within Official Testing, Part I, presentation at DGLR symposium "The MRCA Program TORNADO" 1979)

/5/ R.K. Otnes and L. Enochson, Digital Time Series Analysis
John Wiley and Sons, New York 1972

/6/ J.S. Bendat, A.G. Piersol: Random Data
John Wiley and Sons, New York 1971.

/7/ Smith, Ralph H., A Theory of Longitudinal Short-Period Pilot Induced Oscillations, Interim Report, AFFDL-TR-77, Dayton, Ohio June 1977

/8/ D'Azzo and Houpis, Feedback Control System Analysis and Syntheses,
Mc Graw-Hill Book Company, New York 1966

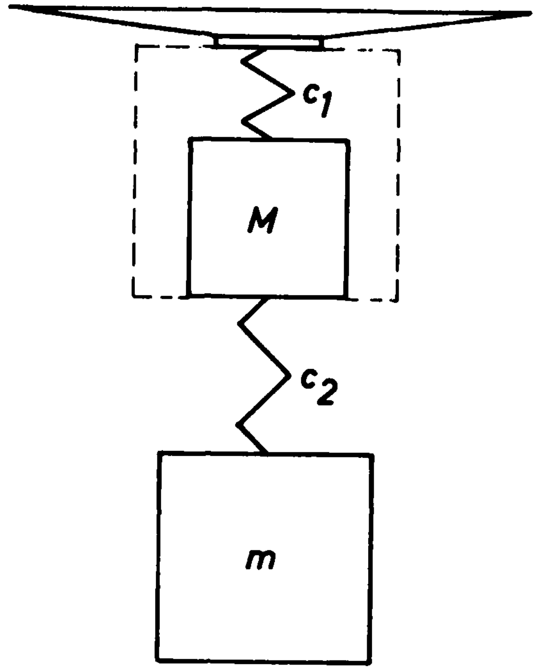


Fig. 1 Model used

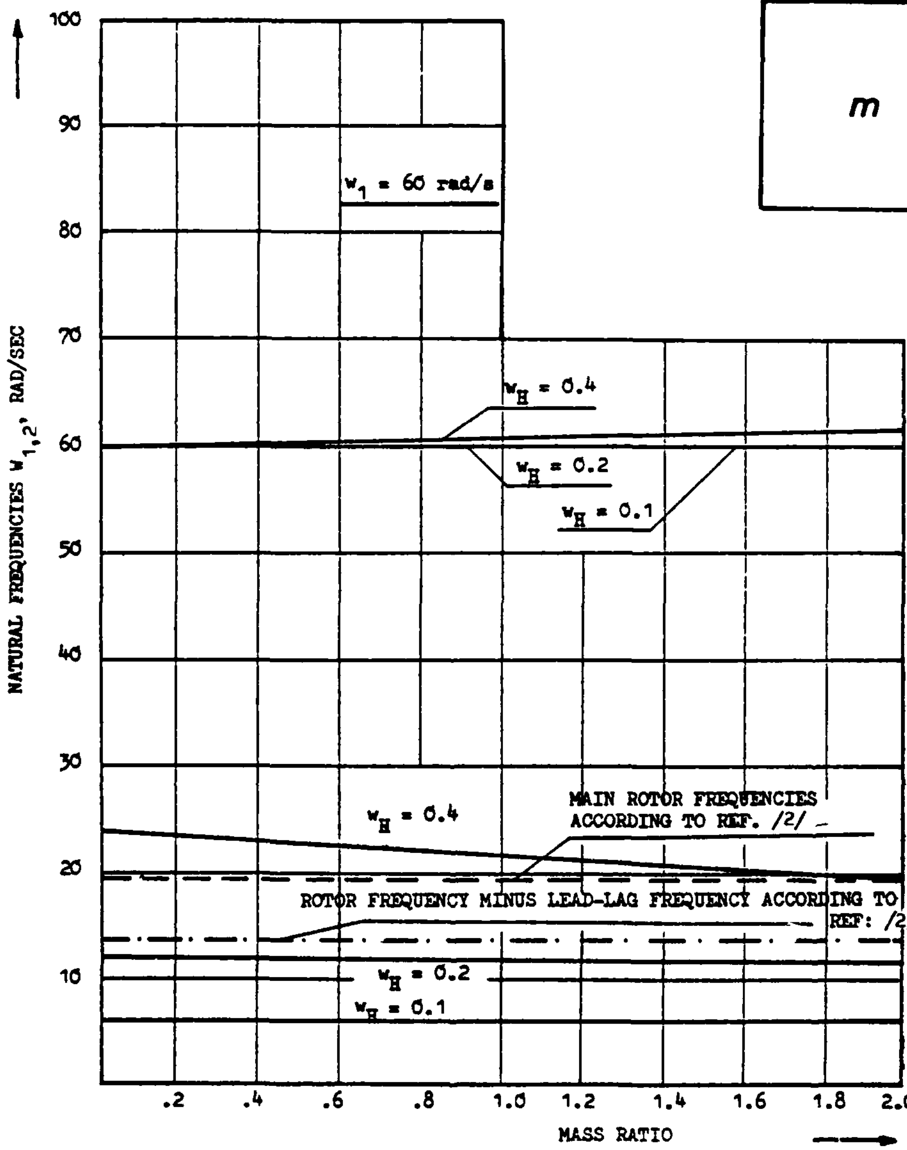


Fig. 2 Natural Frequencies at Vertical Oscillations of the Cargo

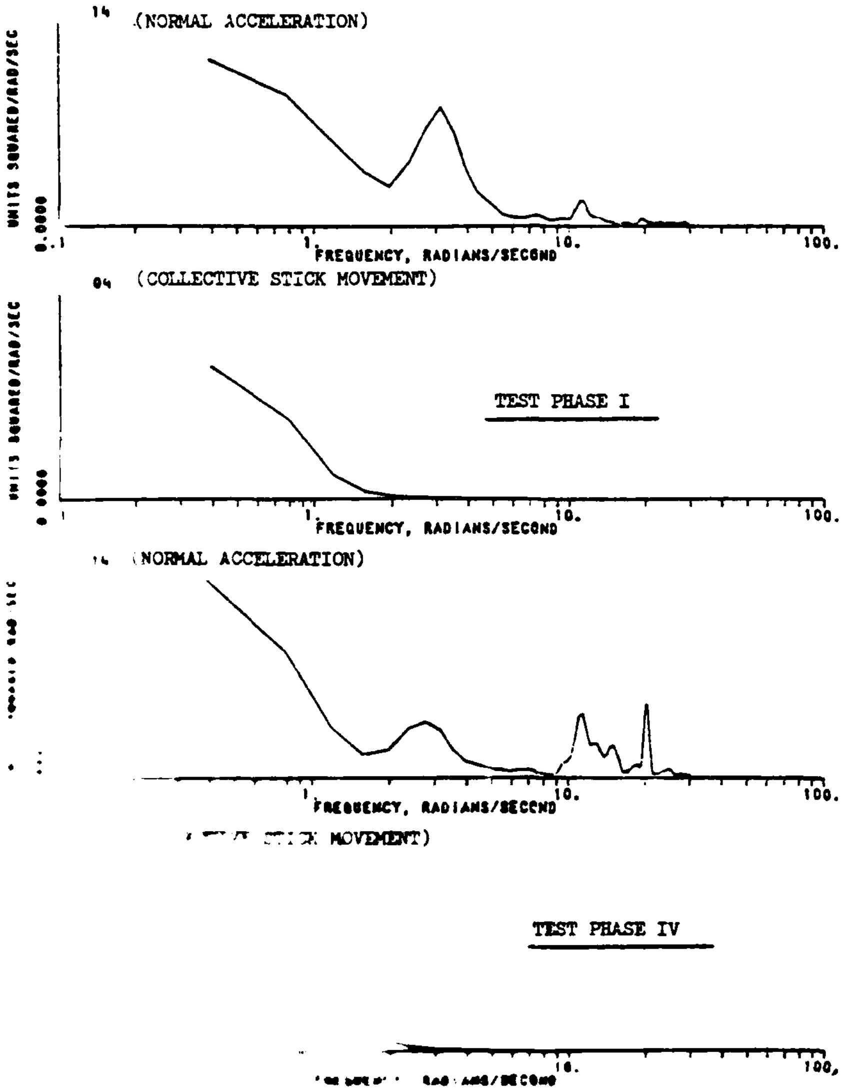
Parameter No.	Parameter Designation	Range	Resolution in Percent of Full-Scale Deflection	Corner Frequency, Hz	Type of Recording
04	Collective Stick Position	0 to 35°	0.1	-	FM-multiplex
04.2	Collective Stick Position by AFCS *)	0 to 30°	0.1	-	FM-direct
05	Pitch Stick Position	0 to 28°	0.1	-	FM-multiplex
05.2	Pitch Stick Position by AFCS *)	0 to 28°	0.1	-	FM-direct
06	Roll Stick Position	0 to 22°	0.1	-	FM-multiplex
06.2	Roll Stick Position by AFCS *)	0 to 22°	0.1	-	FM-direct
14	Normal Acceleration below Pilot Seat	± 4 g	0.5	36	FM-multiplex
17	Roll Rate	± 15°/sec	0.7	6	FM-direct
18	Pitch Rate	± 30°/sec	0.7	15	FM-direct
	Time Code		0.001 sec		FM-direct

*) AFCS = Automatic Flight Control System

Fig. 3 List of Parameters Used

Flight No.	Cargo Mass kg	Material of Cargo Sling	Value of Friction at Collective Stick	Pilot	Remarks
14	-	-	high	A	
15	Internal Cargo 3000	-	high	A	
18	3000	Polyamide	high	A	
19	3000	Chain	high	A	
20	5000	Polyamide	high	A	
21	5000	Chain	high	A	
00	2100	Polyamide	high	B	PIO occurrence during hover
24	6600	Polyamide	high	A	
25	3000	Polyamide	low	A	
26	3000	Polyamide	low	B	
27	5000	Polyamide	low	B	
28	5000	Polyamide	low	A	
29	6600	Polyamide	low	B	
30	6600	Polyamide	low	A	
31	6600	Chain	low	A	PIO occurrence during hover and emergency drop of cargo

Fig. 4 Test Flights Carried Through



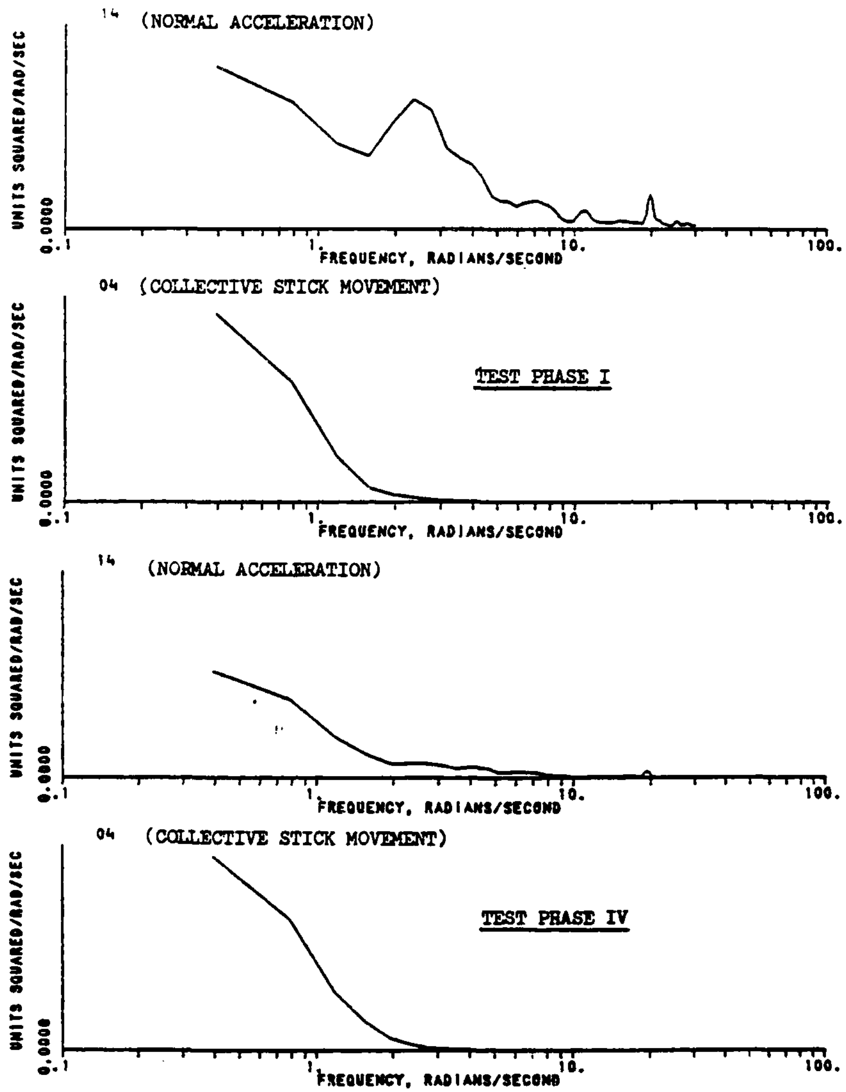
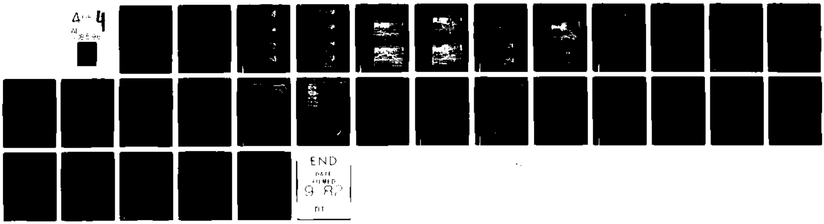
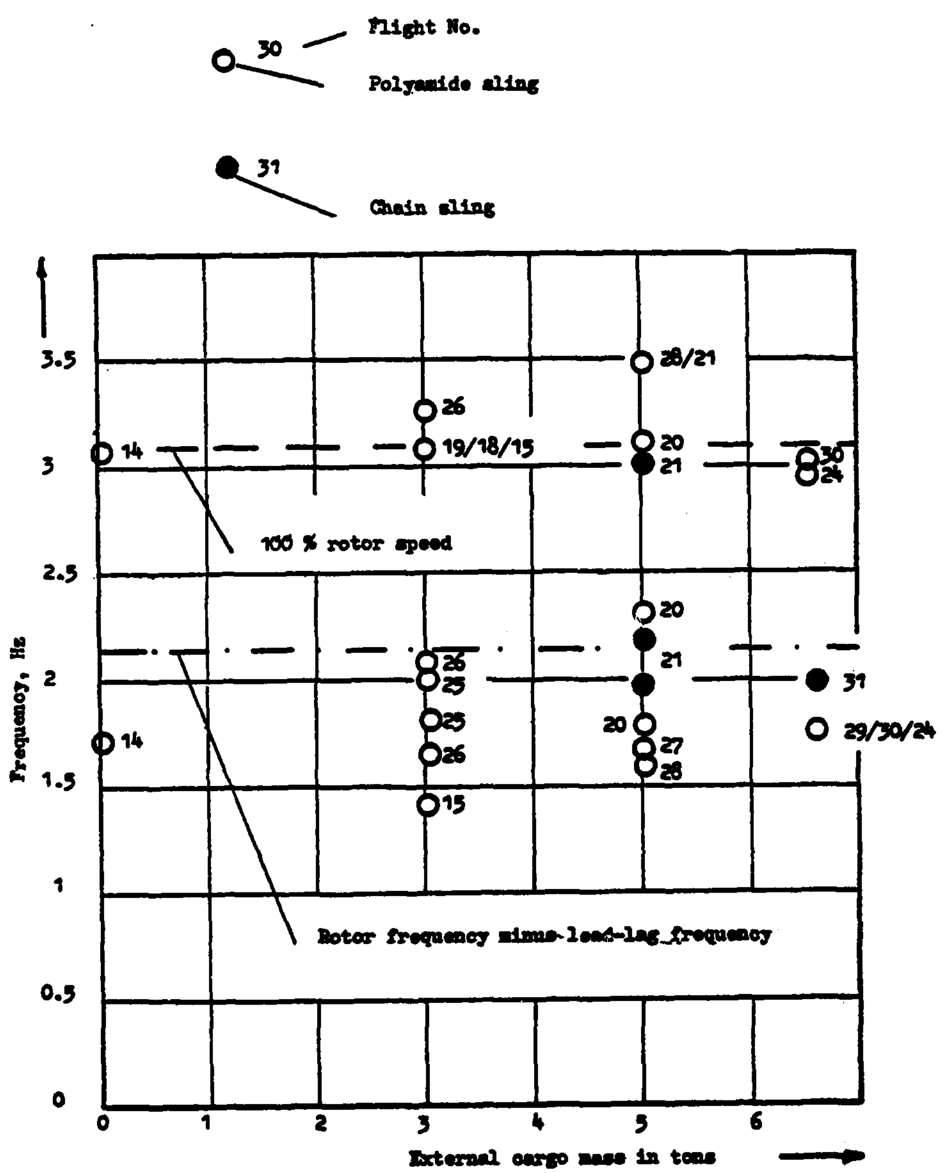


Fig. 5 Power Spectrum Densities of Collective Stick Movement and Normal Acceleration at the Pilot Seat for Test Phases I and IV from Flight 14 (without cargo)

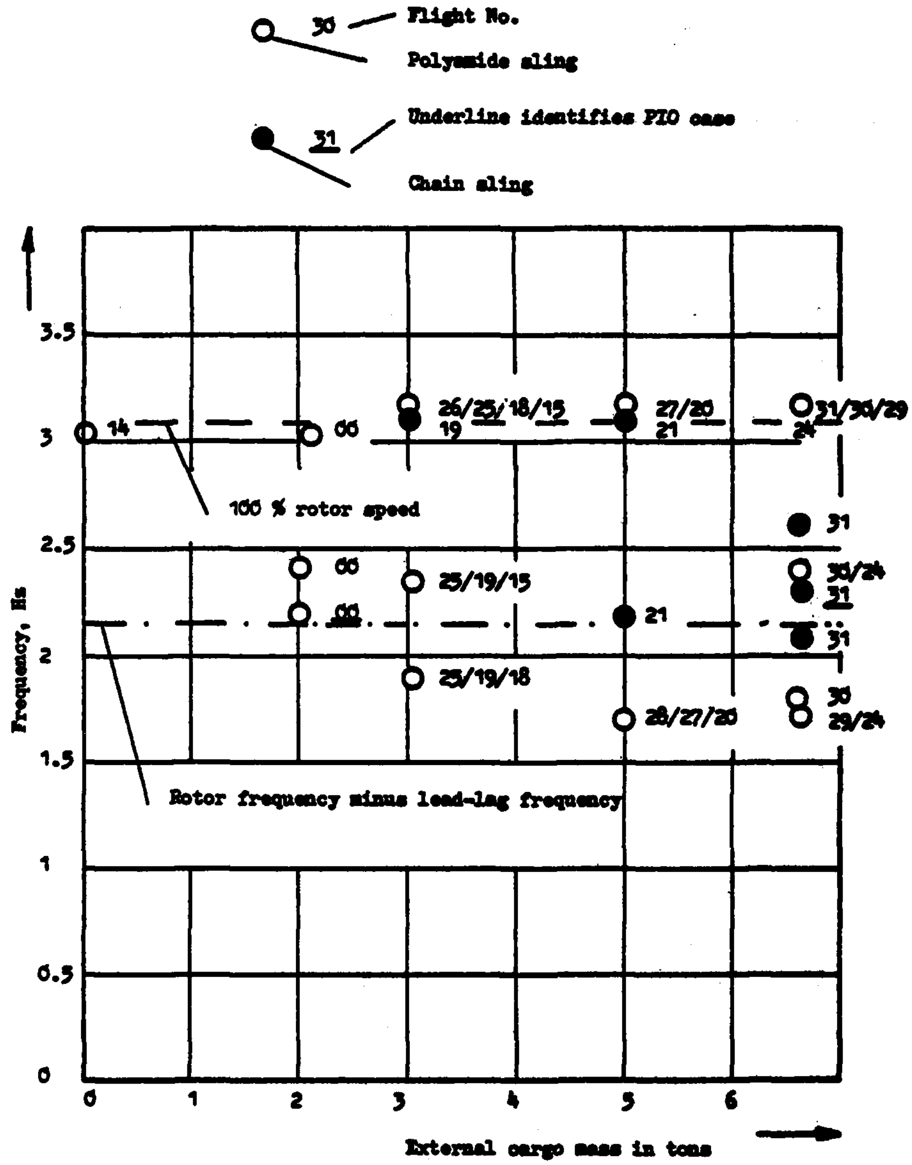
AD-A118 596 ADVISORY GROUP FOR AEROSPACE RESEARCH AND DEVELOPMENT--ETC F/G 1/3
CRITERIA FOR HANDLING QUALITIES OF MILITARY AIRCRAFT.(U)
JUN 82
UNCLASSIFIED AGARD-CP-333 NL





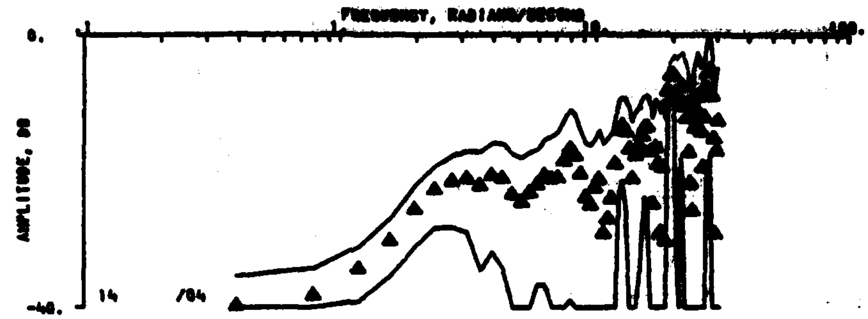
Analysis for test phase I

Fig. 7 Frequency Peaks of the Power Spectral Densities of Normal Acceleration at the Pilot Seat in the Range from 1.4 to 3.5 Hz

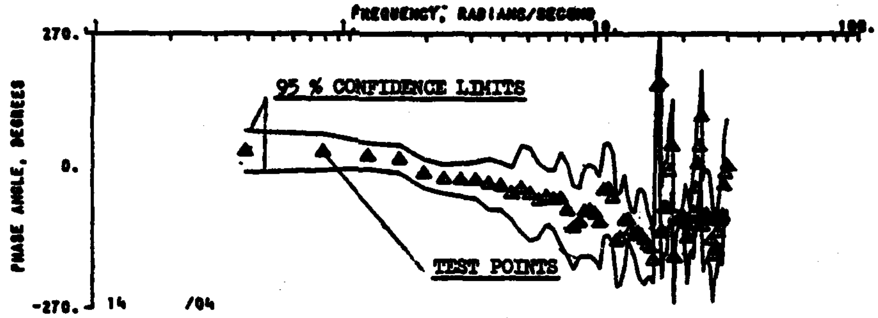


Analysis for test phase IV

Fig. 8 Frequency Peaks of the Power Spectrum Densities of Normal Acceleration of the Pilot Seat in the Range from 1.4 to 3.5 Hz

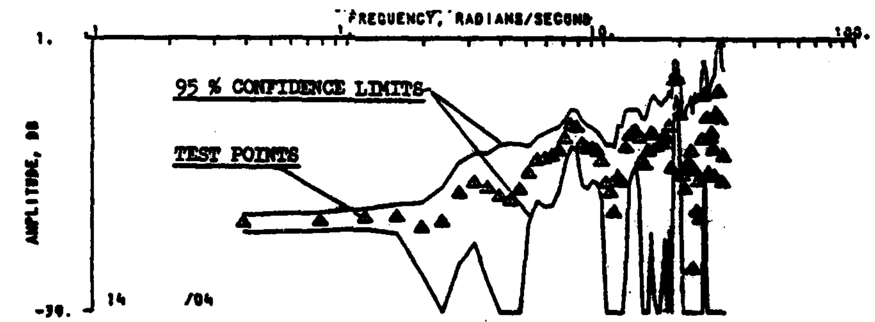


AMPLITUDE RESPONSE

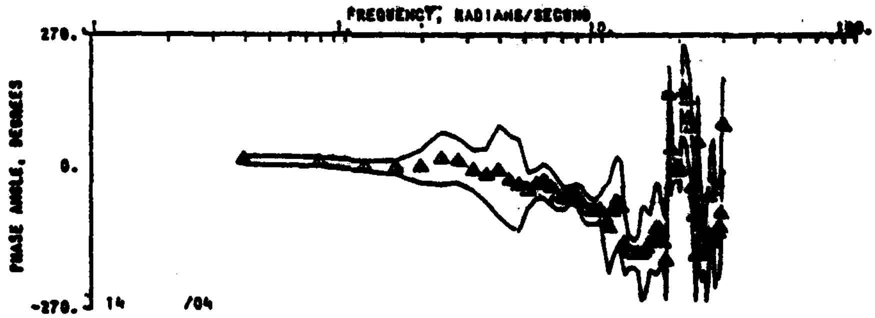


PHASE RESPONSE

Fig. 9 Transfer Function of "Normal Acceleration at the Pilot Seat (14) due to Collective Stick Movement (04)" Flight 14, Test Phase I



AMPLITUDE RESPONSE



PHASE RESPONSE

Fig. 10 Transfer Function "Normal Acceleration at the Pilot Seat (14) due to Collective Stick Movement (04)", Flight 14, Test Phase IV.

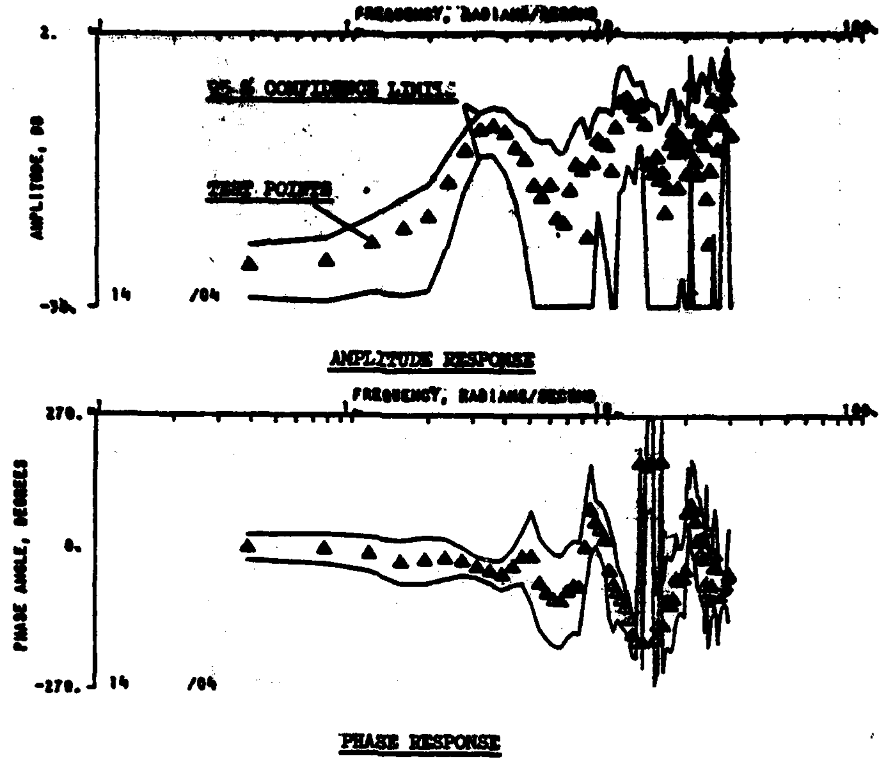


Fig. 11 Transfer Function "Normal Acceleration at the Pilot Seat (14) due to Collective Stick Movement (04)", Flight 30, Test Phase I

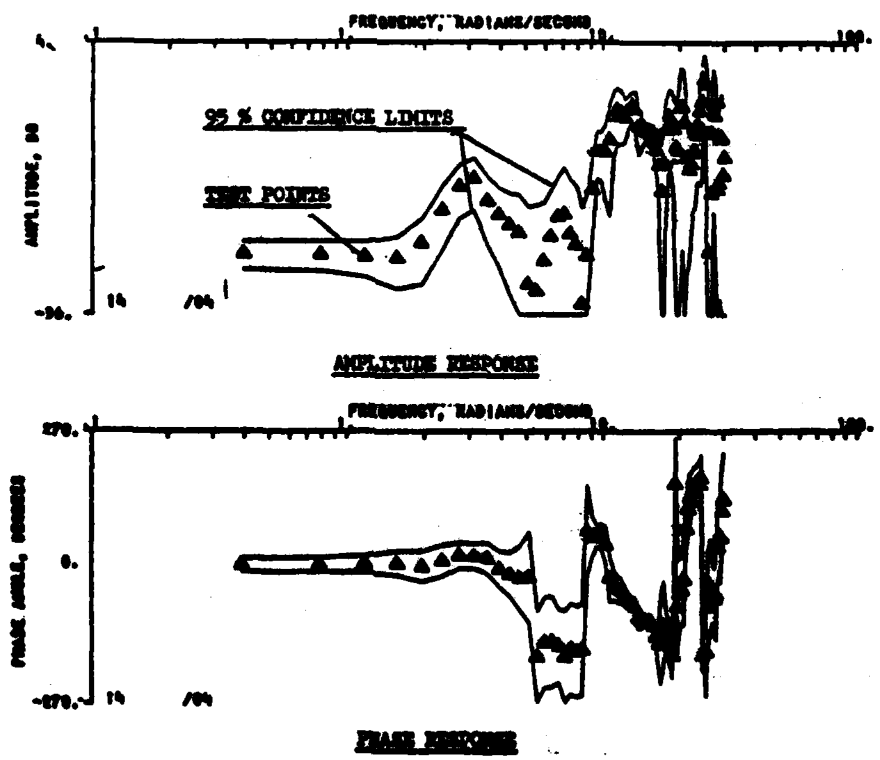
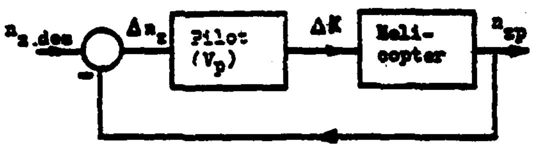


Fig. 12 Transfer Function "Normal Acceleration at the Pilot Seat (14) due to Collective Stick Movement (04)", Flight 30, Test Phase IV



$V_p = K_p \cdot e^{-j\omega t_t}$

K_p = 5db

ω frequency, rad/sec

t_t = 0.2 sec

$n_{z\,des}$	desired normal acceleration
n_{sp}	true -"-
$\Delta n_z = n_{z\,des} - n_{sp}$	
ΔK	collective stick movement

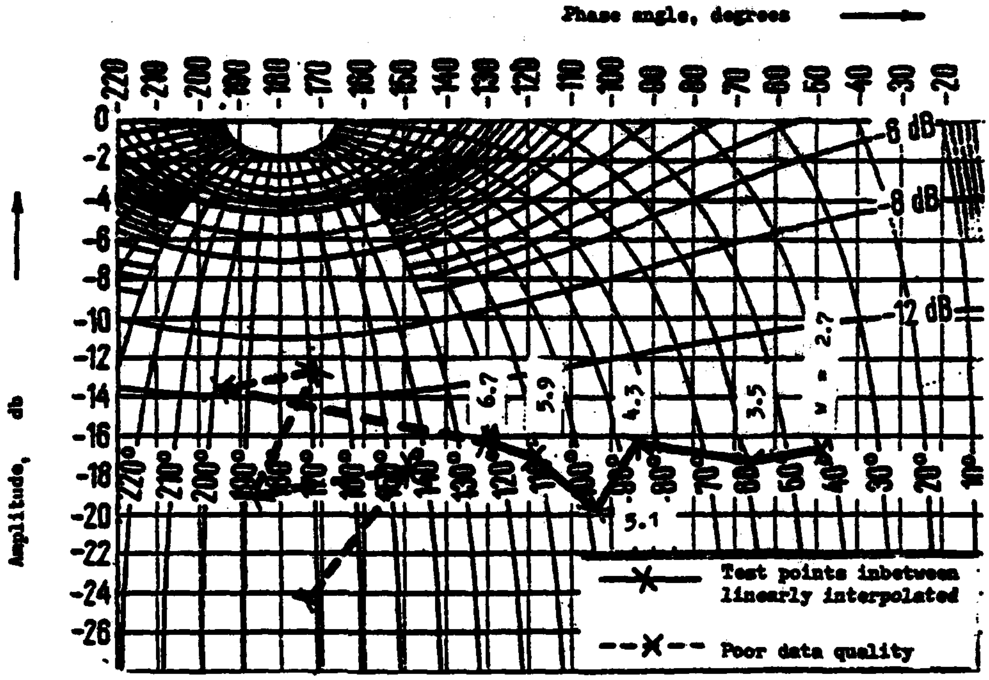


Fig. 13 Nichols Chart for the Transfer Function "n_{sp} due to Δn_z" from Flight 14, Test Phase I

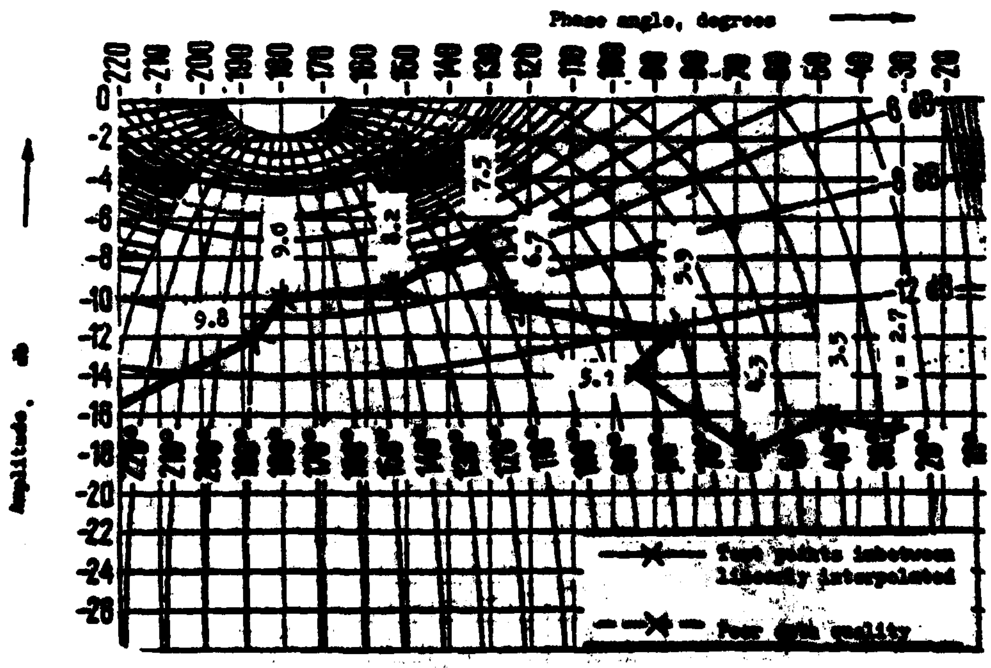
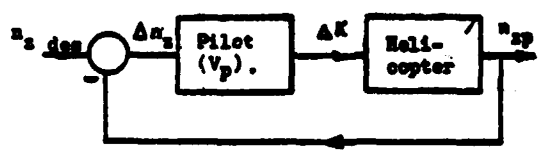


Fig. 14 Nichols Chart for the Transfer Function "n_{sp} due to Δn_z" from Flight 14, Test Phase IV



$V_p = K_p \cdot e^{-j w t_t}$

K_p = 5db

w frequency, rad/sec

t_t = 0.2 sec

n_{zdes} desired normal acceleration

n_{zp} true -"-

ΔK collective stick movement

$\Delta n_z = n_{z\,des} - n_{zp}$

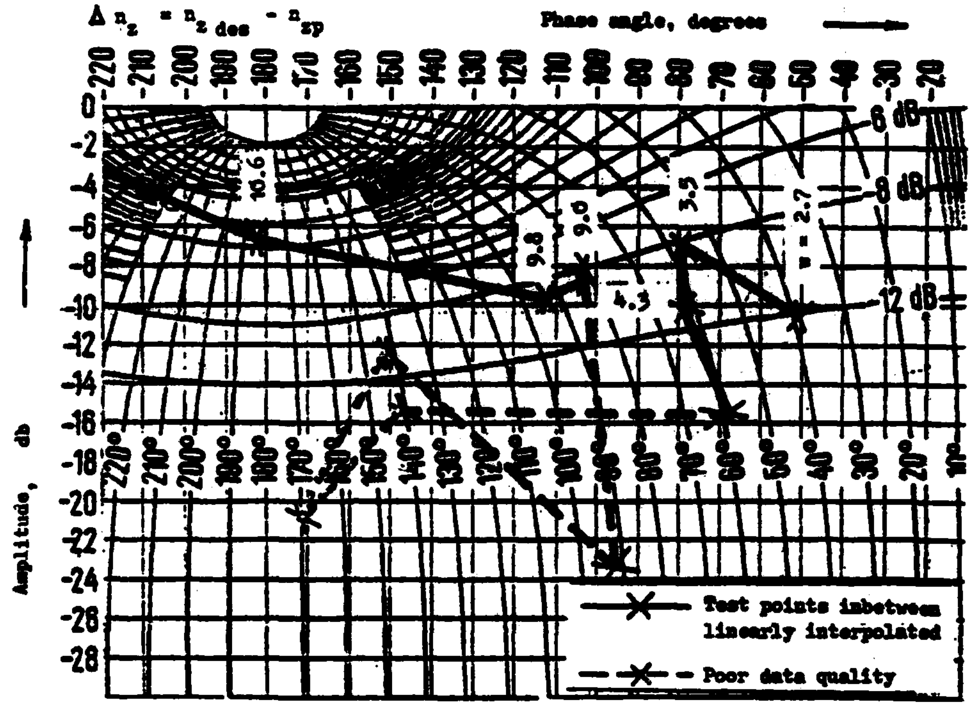


Fig. 15 Nichols Chart for the Transfer Function "n_{zp} due to Δn_z" from Flight 30, Test Phase I

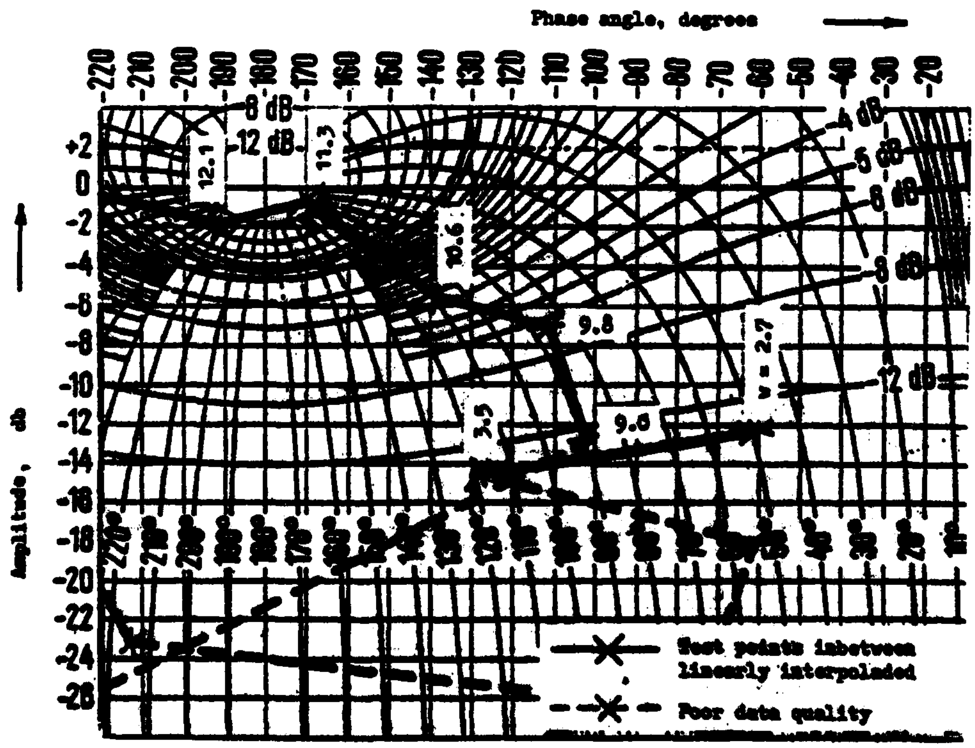


Fig. 16 Nichols Chart for the Transfer Function "n_{zp} due to Δn_z" from Flight 30, Test Phase IV

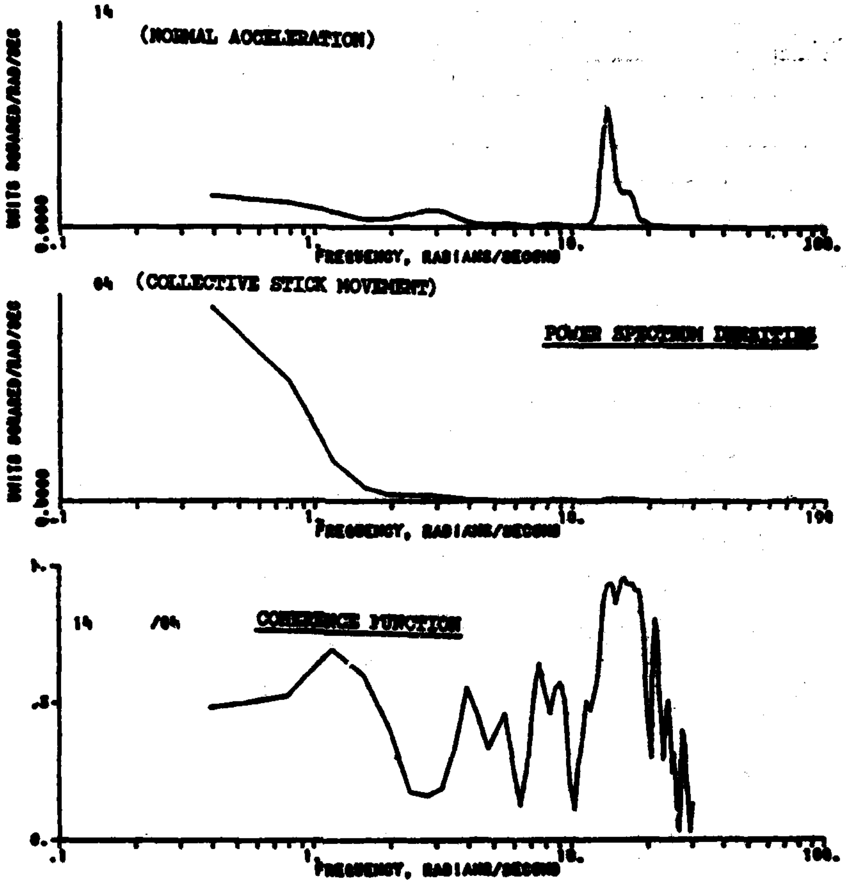


Fig. 17 Power Spectrum Densities of Collective Stick Movements and Normal Acceleration at the Pilot Seat with Associated Coherence Function from Flight 31, Test Phase Leading to PIO and Cargo Drop.

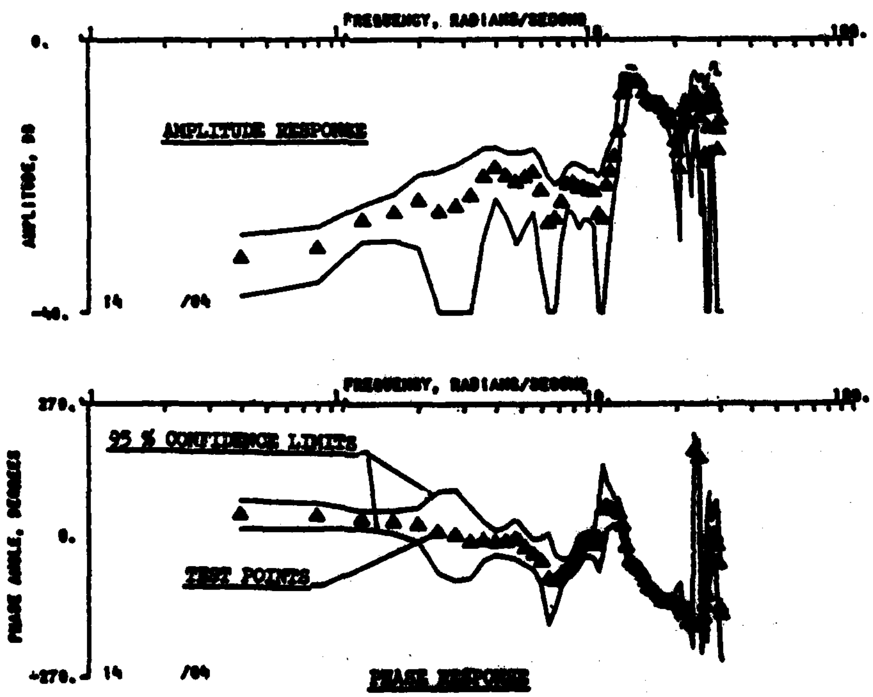
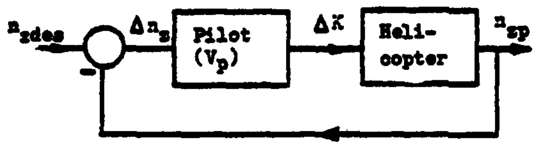


Fig. 18 Transfer Function "Normal Acceleration at the Pilot Seat (14) due to Collective Stick Movement (64) from Flight 31, Test Phase Leading to PIO and Cargo Drop



$V_p = K_p \cdot e^{-jwt_t}$

$K_p = 5\text{db}$

w frequency, rad/sec

$t_t = 0.2$ sec

$n_{z\,des}$ desired normal acceleration

n_{zp} true -"-

$\Delta n_z = n_{z\,des} - n_{zp}$

ΔK collective stick movement

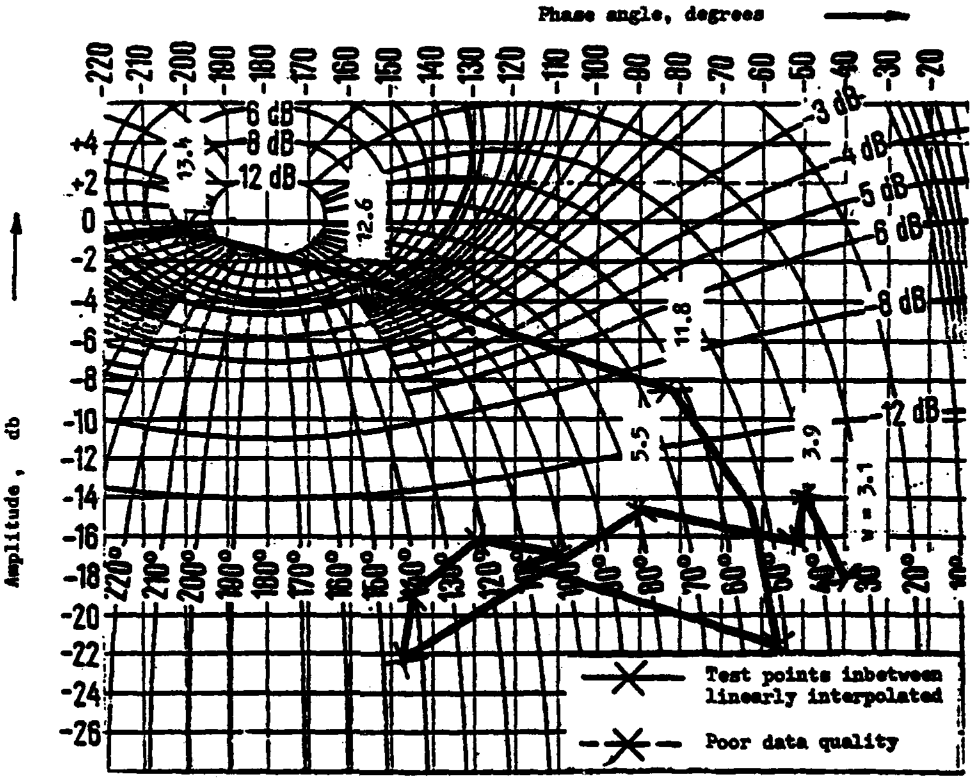


Fig. 19 Nichols Chart for the Transfer Function "n_{zp} due to Δn_z" from Flight 31, Test Phase Leading to PIO and Cargo Drop.

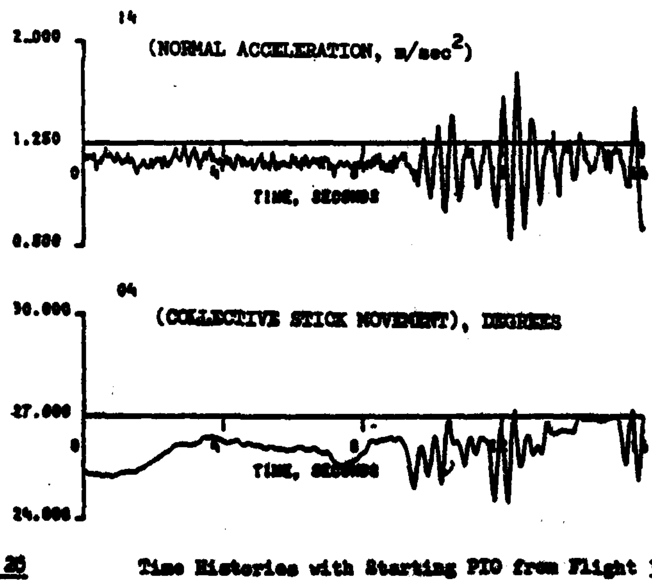


Fig. 20 Time Histories with Starting PIO from Flight 31

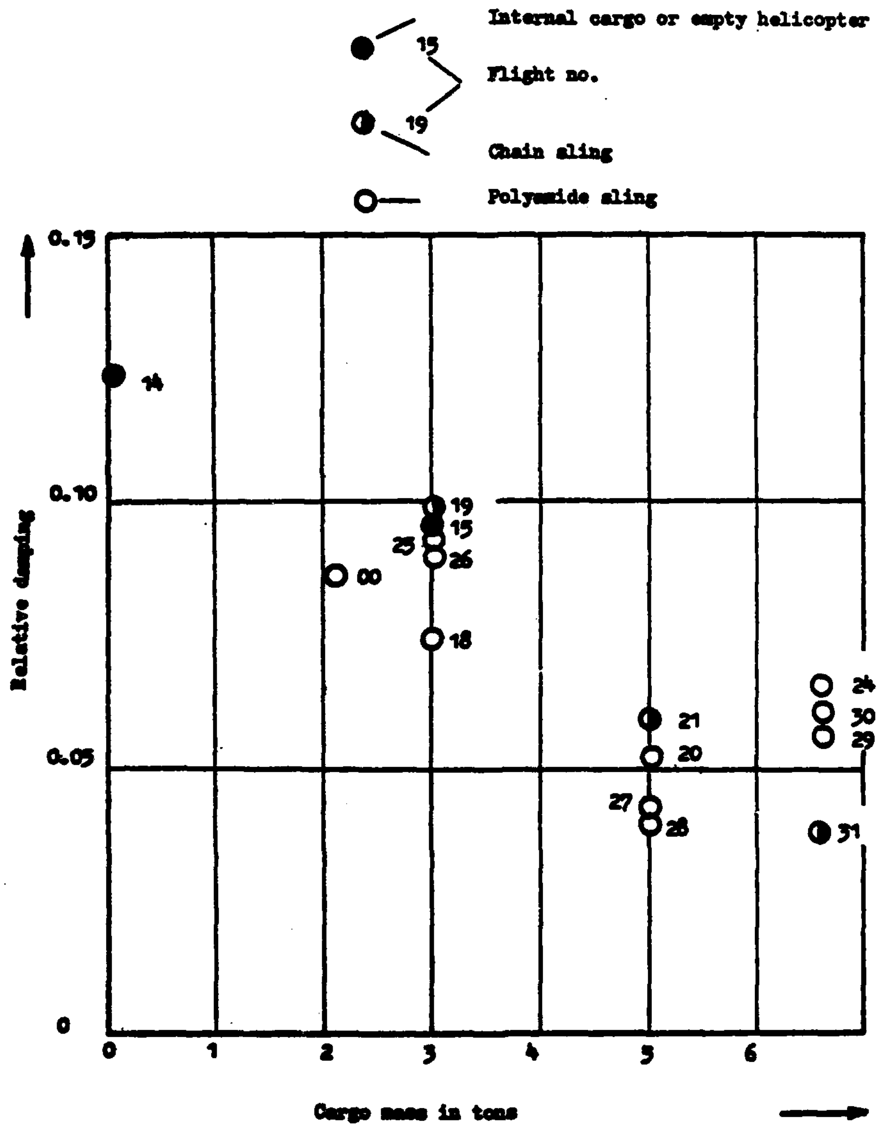
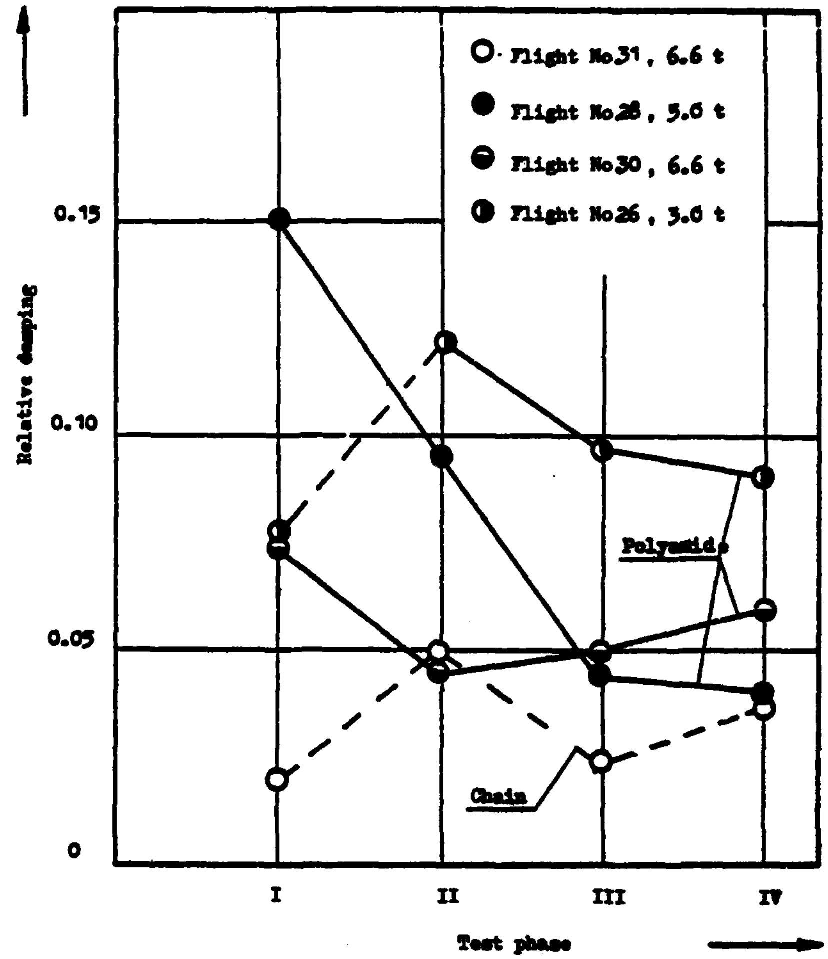


Fig. 22 Dependence of Relative Damping on the Cargo Mass for Test Phase IV



Test phase
I: Flight including turns
II: Acceleration - deceleration
III: Landing approach
IV: Hover

Fig. 21 Relative Damping Depending on Flight Test Phase and Cargo Mass

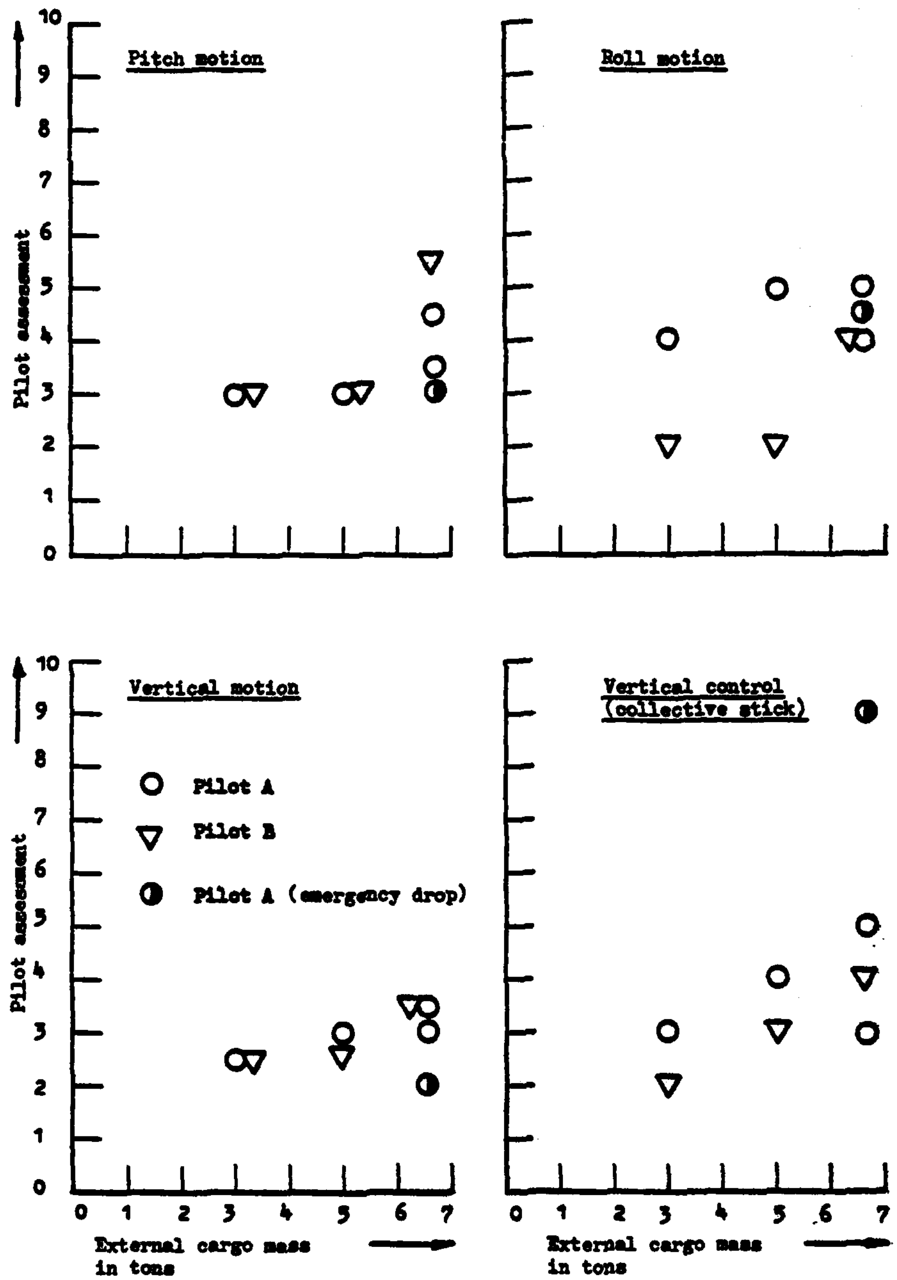


Fig. 23 Summary of Pilot Assessment Taken from the Assessment Forms.

PREDICTION OF AIRCRAFT HANDLING QUALITIES USING ANALYTICAL MODELS OF THE HUMAN PILOT

by
Ronald A. Hess
Research Scientist
NASA Ames Research Center,
Moffett Field, California 94035

SUMMARY

The optimal control model (OCM) of the human pilot is applied to the study of aircraft handling qualities. Attention is focused primarily on longitudinal tasks. The modeling technique differs from previous applications of the OCM in that considerable effort is expended in simplifying the pilot/vehicle analysis. After briefly reviewing the OCM, a technique for modeling the pilot controlling higher order systems is introduced. Following this, a simple criterion for determining the susceptibility of an aircraft to pilot-induced oscillations (PIO) is formulated. Finally, a model-based metric for pilot rating prediction is discussed. The resulting modeling procedure provides a relatively simple, yet unified approach to the study of a variety of handling qualities problems.

1. INTRODUCTION

The advent of modern digital stability and control augmentation systems has created a renewed interest in the study of aircraft longitudinal handling qualities. This renewed interest is attributable to two factors: First, the higher order nature of the dynamics typically associated with digital control systems makes analytical prediction of handling qualities difficult. Contemporary handling qualities specifications (Ref. 1) are written assuming "classical" aircraft characteristics, e.g., in the longitudinal mode, the existence of distinct and dominant short-period dynamics is assumed. With modern systems, the short-period characteristics may be dramatically altered by feedback and the higher order control system dynamics may dominate the vehicle handling qualities. Second, shortcomings in predictive techniques are made even more critical by the fact that severe handling qualities deficiencies often arise in practice which are directly attributable to the higher order nature of the digital control law implementation. An example of this is the ability of high frequency phase lags or time delays in the control system to sharply degrade aircraft handling qualities and to be a contributing factor to pilot-induced oscillations (Ref. 2).

In the research to be described, a pilot-modeling technique for handling qualities research, discussed in Ref. 3, is utilized and extended to cover higher order systems. The characteristics of over thirty aircraft configurations are analyzed, primarily in the longitudinal mode. Particular emphasis is placed upon those configurations where control system dynamics and time delays have been recognized as contributing factors to handling qualities deficiencies. The contribution of vehicle/control system dynamics to PIO tendencies is outlined, and a metric for pilot rating prediction is discussed.

2. BACKGROUND

The pilot-modeling technique as discussed in Ref. 3 forms the framework for the research described here. This technique utilizes the optimal control model of the human pilot and a novel method for the a priori selection of dominant OCM parameters (index of performance weighting coefficients and observation noise/signal ratios). A brief tutorial review of the procedure for selecting index of performance weighting coefficients is now presented. Consider the longitudinal tracking task of Fig. 1 in which the pilot is attempting to minimize pitch attitude deviations $\theta(t)$ in the presence of atmospheric disturbances. Ignore the dashed "internal attitude command" for the present. An acceptable index of performance for this task would be (Ref. 3)

$$J = E\left\{\lim_{X \to \infty} \frac{1}{2X} \int_{-X}^{X} \left[\theta^2(t)/\theta_M^2 + \dot{\delta}^2(t)/\dot{\delta}_M^2\right]dt\right\}$$

where $\dot{\delta}(t)$ is control rate.

As discussed in Ref. 3, we assign an arbitrary maximum allowable deviation to the time rate of change of the error, $\dot{\theta}(t)$, and denote it $\dot{\theta}_M$. Now an effective time constant T can be introduced to define maximum allowable deviations of the integral and derivatives of $\dot{\theta}(t)$ as:

$$\theta_M = \dot{\theta}_M T \quad ;$$

$$\dot{\theta}_M = \text{specified but arbitrary} \quad ;$$

$$\ddot{\theta}_M = \dot{\theta}_M / T \quad ; \tag{1}$$

and

$$\dddot{\theta}_M = \ddot{\theta}_M / T = \dot{\theta}_M / T^2 \quad .$$

$$\vdots$$

The justification for using a single time constant to represent the ratio of the maximum value of a variable to that of its next highest derivative rests upon the system bandwidth implications which follow when Eq. (1) is used in implementing the OCM. We will also assign a maximum allowable deviation to the time rate of change of the pilot's control, $\dot{\delta}(t)$, and denote it $\dot{\delta}_M$. Similar to Eq. (1) we write

$$\delta_M = \dot{\delta}_M T \quad ;$$

$$\dot{\delta}_M = \text{to be selected} \quad ;$$

$$\ddot{\delta}_M = \dot{\delta}_M / T \quad ;$$

$$\dddot{\delta}_M = \ddot{\delta}_M / T = \dot{\delta}_M / T^2 \quad ; \tag{2}$$

$$\vdots$$

The value of $\dot{\delta}_M$ is not arbitrary, however, but is found using Eqs. (1) and (2) and the vehicle dynamics as follows: Let the pitch attitude dynamics of the aircraft be given by

$$\frac{\theta}{\delta}(s) = K \frac{s^{n-1} + a_{n-2}s^{n-2} + \ldots + a_1 s + a_o}{s^n + b_{n-1}s^{n-1} + \ldots + b_1 s + b_o} \quad . \tag{3}$$

Then, as explained in Ref. 3, we write

$$\dot{\delta}_M = \frac{1/T^{n-1} + |b_{n-1}|/T^{n-2} + \ldots + |b_1| + |b_o|T}{K\left(1/T^{n-2} + |a_{n-2}|/T^{n-3} + \ldots + |a_1| + |a_o|T\right)} \dot{\theta}_M \tag{4}$$

Thus, once T is known, $\dot{\delta}_M$ and θ_M (and, if needed, $\ddot{\theta}_M$, etc.) can be determined immediately. Choosing T involves selecting a domain of $1/T$: $1/4\tau < 1/T < 4/\tau$ and then plotting J, the value of the OCM index of performance, vs $1/T$. The operating point or "knee" of this curve determines T. The knee is defined as the point where

$$\frac{\partial J}{\partial \log(1/T)} = \eta_\delta \frac{J|_{T=\tau/4} - J|_{T=4\tau}}{\log(4/\tau) - \log(1/4\tau)} \quad . \tag{5}$$

Here η_δ is a constant, nominally unity, which can be used to reflect manipulator characteristics, much like an efficiency factor; τ is the pilot's time delay (nominally 0.2 sec). $J|_{T=\tau/4}$ is the value of the index of performance which results when $T = \tau/4$.

The ability of the OCM parameter selection technique to provide a pilot model which matches measured pilot describing functions, remnant power spectral densities and root mean square (RMS) performance measures was demonstrated in Ref. 3. In addition, the

modeling technique was shown capable of providing qualitative and quantitative handling qualities assessments. The method for selecting observation noise/signal ratios for the OCM is discussed in Ref. 3 and will not be dealt with here.

Although Eq. (3) shows dynamics of arbitrary order; all the pitch attitude dynamics of Ref. 3 were of the form:

$$\frac{\theta}{\delta} = \frac{K_\theta(s + 1/T_L)}{s(s^2 + 2\zeta_n\omega_n s + \omega_n^2)} \quad . \tag{6}$$

When higher order dynamics are encountered, the method for selecting the operating point needs to be modified slightly. The large phase lags typically associated with the dynamics of vehicles with higher order dynamics need to be reflected in choosing the domain of $1/T$ to be used in Eq. (5). To accomplish this, a delay τ_D is defined as the delay which accrues when the vehicle dynamics of Eq. (3) are represented as

$$\frac{\theta}{\delta} = \frac{K_\theta(s + 1/T_L)e^{-\tau_D s}}{s(s^2 + 2\zeta_n\omega_n s + \omega_n^2)} \quad . \tag{7}$$

The parameters on the right hand side of Eq. (7) are found using a program to fit a linear transfer function model to the actual vehicle dynamics (Ref. 4). Equation (5) is modified by simply replacing τ with $\tau + \tau_D$. The resulting equation is interpreted graphically in Fig. 2. Calculating τ_D and including it in Eq. (5) constitutes the extension of the methods of Ref. 3 to higher order systems. It is important to emphasize that the actual higher order vehicle dynamics are used in the modeling procedure; Eq. (7) is employed *only* to select τ_D which, in turn, determines the domain of $1/T$ used in finding the index of performance weighting coefficients.

3. APPLICATION TO AIRCRAFT HANDLING QUALITIES

3.1 Pilot-Induced Oscillations

Table I lists the aircraft configurations which have been analyzed in this study. The designations in the column labeled "Configuration" use notation found in the corresponding references. The first sixteen deal with high performance fighter-type aircraft in tracking or landing approach conditions and are taken from Refs. 2, 5, and 6. These configurations constitute the test cases for the majority of the assessments. The next four configurations are taken from Ref. 7 and represent pilot-in-the-loop simulations of a hovering helicopter. Configurations 21-25 are flight test results from Ref. 9 in which the Princeton University Variable Response Aircraft (VRA) was used to determine the effect of digital sampling rates and time delays on longitudinal handling qualities. The vehicle dynamics appropriate for 105 kts airspeed were used in the modeling procedure. The pilot ratings used were average values obtained from altitude tracking and approach and landing tasks (Fig. 3 of Ref. 9). Finally, configurations 26-32 are taken from Ref. 10 where a moving-base simulator experiment on the NASA Ames Flight Simulator for Advanced Aircraft (FSAA) was described which investigated a wings level-turn control mode for air-to-ground weapons delivery. Note that unlike the previous twenty-five configurations, these involve lateral-directional aircraft handling qualities. The effective vehicle dynamics for the lateral gunsight aiming task were parameterized by a damping ratio ζ_n, an undamped natural frequency ω_n and a pure time delay τ_D (Ref. 10). The data for the so-called "fine" task were used. This task is explained in Ref. 10.

As an example of the modeling results, Fig. 3 shows the longitudinal open-loop pilot/vehicle characteristics (Y_pY_c) for three of the configurations used in Ref. 2. Here, the NASA Dryden F-8 digital fly-by-wire aircraft is considered with a rudimentary augmentation system ("Pitch Direct") and three transport time delays of 0.13 sec, 0.23 sec, and 0.33 sec, respectively. The predicted effect of the time delays is apparent in the reduced open-loop crossover frequencies ω_c. This open-loop characteristic obviously has a deleterious effect on the closed loop θ/θ_c transfer functions as shown in Fig. 4 [$\theta/\theta_c = Y_pY_c/(1 + Y_pY_c)$]. This transfer function is important in assessing PIO susceptibility. Although the task has been defined as pitch-attitude disturbance regulation, attitude commands θ_c internally generated by the pilot would be employed in precise *altitude* regulation (dashed line in Fig. 1). Note in Fig. 4, that as τ_D increases, $|\theta/\theta_c|$ and $\angle\theta/\theta_c$ decrease at all frequencies. Perfect command following, of course, implies $\theta/\theta_c = 1.0$ at all frequencies. In Fig. 4, $|\theta/\theta_c| < 1.0$ for all configurations when $\omega < 3.0$ rad/sec, and is particularly poor for the configuration with $\tau_D = 0.33$ sec. It can be readily shown that open-loop crossover frequencies less than 3-4 rad/sec will invariably result in poor closed-loop attitude command-following characteristics. The simplest and most direct way for the pilot to attempt to improve this closed loop command-following performance is to increase ω_c by increasing his

static gain. If the pilot attempts this for the F-8 configuration with $\tau_D = 0.33$ sec, a very lightly damped closed-loop oscillation occurs at $\omega = 3.3$ rad/sec (see Fig. 4). This is identical to the PIO frequency shown in Ref. 2 for this configuration.

Similar results are also obtained for configurations from Ref. 5. Figure 5 compares a pair of open-loop transfer functions obtained using configurations "11" and "12" from Ref. 5 and applying the pilot-modeling technique discussed above. Once again, the dramatic difference in the crossover frequencies ω_c is apparent. The effects of the pilot's attempting to improve the performance of configuration "12" by increasing his static gain by 10 dB are shown in Fig. 6. Once again, a lightly damped oscillatory mode is seen to appear. The simulations of Ref. 5 were intended to provide performance comparisons for configurations which were flight tested and discussed in Ref. 8. The latter report included Pilot-Induced-Oscillation-Ratings (PIOR) obtained using the scale of Fig. 7. It is interesting to note that configuration "11" received an average PIOR of 1 indicating a very satisfactory vehicle whereas configuration "12" received a marginal average rating of 2.7 indicating a vehicle with definite PIO tendencies. These experimental results are seen to corroborate the analytical findings just discussed.

Next, consider two configurations from Ref. 6 denoted as "4-1" and "6-1." Figure 8 shows the Y_pY_c plots for these configurations. Configuration "4-1" received a very satisfactory PIOR of 1 whereas configuration "6-1" received a very poor PIOR of 4. Indeed, configuration "6-1" produced a PIO in flight test with a frequency of approximately 3.75 rad/sec. Analytically increasing the pilot's static gain by 4.75 dB (the limit for closed-loop stability) in the modeling-results for this configuration produced a closed-loop oscillation at approximately 3.5 rad/sec. This 4.75 dB increase would increase ω_c from around 1.5 rad/sec to only around 2.5 rad/sec as compared to a value of 4.5 rad/sec for configuration "4-1."

Figure 9 shows the predicted Y_pY_c's for a pair of configurations from Ref. 9. The task was longitudinal control in approach and landing using the Princeton VRA. The variable of interest here was the amount of effective delay in the control system. In the first, an effective delay of 0.055 sec was employed, while in the second, 0.355 sec was used. Again, note the striking difference in crossover frequencies in the predicted pilot/vehicle dynamics. In the first case, $\omega_c = 3.4$ rad/sec, while in the latter, $\omega_c = 0.55$ rad/sec. Flight test of the first configuration showed no PIO tendencies, while those for the latter produced PIO's (Ref. 9).

Finally, Fig. 10 shows the predicted Y_pY_c's for a pair of configurations from Ref. 10. In the first, the control system parameters were $\zeta_n = 1.4$, $\omega_n = 2.0$ rad/sec and $\tau_D = 0.0$ sec, while in the second, $\zeta_n = 1.4$, $\omega_n = 15.0$ rad/sec and $\tau_D = 0.49$ sec. The ω_c difference is again evident. Simulation results indicated that the configuration with delay was definitely PIO prone and the one without delay was not. It is interesting to point out that the configuration without delay still received an average Cooper-Harper pilot rating of 6.5, even though it was not PIO prone. Thus, poor pilot ratings, per se, are not a necessary condition for PIO susceptibility.

In each of the cases above, we have made direct comparisons of vehicles which were found to be PIO prone with those which were not. This was done to emphasize the fact that the method proposed here is clearly discriminatory in predicting PIO susceptibility. The simple criterion for exonerating a vehicle from PIO tendencies requires that the predicted pilot/vehicle crossover frequencies associated with inner attitude-loops be greater than 3-4 rad/sec.

3.2 Cooper-Harper Ratings

Figure 11 is a plot of the Cooper-Harper ratings which the thirty-one configurations from Table I received in simulation or flight test vs the value of a proposed handling qualities metric defined as $K_i[(\tau + \tau_D)/\tau]^{1/4} \cdot J$. No ratings were reported in Ref. 10 for configuration 32 of Table I. Hence, only thirty-one data points are shown in Fig. 11. The K_i can be interpreted as a "calibration parameter" which, when multiplied by $[(\tau + \tau_D)/\tau]^{1/4} \cdot J$, allows the reported pilot ratings from different tasks and data sources to coalesce as shown in Fig. 11. Note that we do *not* allow K_i to vary within the analysis of any particular task, regardless of configuration changes. Thus, the analysis of the six configurations from Ref. 2 used a single value of K_i (call it K_1). The analysis of the seven configurations from Refs. 5 and 8 used a single value of K_i (call it K_2), etc. In all, six different K_i values (each one corresponding to the six different symbols in Fig. 11) were used to generate Fig. 11. With the exception of K_i, all the parameters of the metric are an intrinsic part of the modeling procedure, and, as such, involve no guesswork on the part of the analyst. In order to determine K_i, the analyst must have an actual pilot rating for one of the configurations tested for the task under study. If the analyst does not have such a rating available, Fig. 11 is still useful, since the curve is nearly linear from a pilot rating of about 2.0 to 10.0, a range which covers 80% of the Cooper-Harper scale. Thus, *relative* rating changes may be able to be predicted using the linear portion of the curve. Note that, with the exception of one data point (Config. 19 from Ref. 7), the scatter in the ratings in Fig. 11 is only about $\pm\frac{1}{2}$ a pilot rating.

The inclusion of the factor $[(\tau + \tau_D)/\tau]^{1/4}$ in the metric deserves a brief discussion. In previous research with the OCM, the value of J, alone, has been found to correlate well with pilot opinion rating (Ref. 11). In many of the configurations studied here, however (those with $\tau_D > 0$), the value of J was not acceptable as a metric. In

general, the "predicted" opinion rating increments were smaller than those reported in experiment. There appears to be a reason for this based upon pilot tracking performance. Namely, when the task is disturbance regulation involving relatively low-bandwidth turbulence, large time delays are not necessarily a harbinger of dramatic deterioration in tracking performance. This is analytically verified by considering the RMS tracking scores for configurations 1 and 3 from Table I. Here, a 154% increase in time delay between configurations 1 and 3 involves a log ωc regression of nearly a decade. However, the predicted RMS pitch attitude score increases by only 36% and the predicted RMS control-rate score actually decreases. As we have attempted to point out here, however, the same cannot be said for discrete command following or abrupt maneuvers. In this case, ω_c regression can have a significant impact on the ability of the closed-loop pilot/vehicle system to follow abrupt, internally generated commands. It certainly is not unreasonable to postulate that such short-term response characteristics (in addition to RMS characteristics) are reflected in pilot opinion rating. Indeed, recorded pilot comments support this notion (e.g., Refs. 2 and 6). The inclusion of $[(\tau + \tau_D)/\tau]^4$ in the metric appears to account for the influence of these delays on pilot opinion in a straightforward manner, employing an easily identifiable parameter (τ_D).

4. CONCLUSIONS

The research summarized in this paper provides a unified approach to pilot/vehicle analysis, and in particular for:

1) Modeling the pilot controlling higher order systems.

2) Predicting the susceptibility of aircraft to longitudinal PIO's.

3) Predicting pilot ratings for tasks when one configuration rating is known, or predicting relative rating changes between configurations.

Although the majority of tasks studied dealt with longitudinal control, five lateral-directional configurations were successfully analyzed with no changes in the modeling technique.

5. REFERENCES

1. Anon.: Military Specification, Flying Qualities of Piloted Airplanes: MIL-F-8785B(ASG), August 1969.

2. Berry, D.T., Powers, Bruce G., Szalai, K.J., and Wilson, R.J.: A Summary of an In-Flight Evaluation of Control System Pure Time Delays During Landing Using the F-8 DFBW Airplane, AIAA Paper No. 80-1626. 1980 AIAA Atmospheric Flight Mechanics Conference, Danvers, Mass.

3. Hess, R.A.: A Pilot Modeling Technique for Handling Qualities Research, AIAA Paper No. 80-1624. 1980 AIAA Atmospheric Flight Mechanics Conference, Danvers, Mass.

4. Seidel, R.C.: Transfer-Function-Parameter Estimation From Frequency Response Data--A FORTRAN Program. NASA TM X-3286, 1975.

5. Arnold, J.D.: An Improved Method of Predicting Aircraft Longitudinal Handling Qualities Based on the Minimum Pilot Rating Concept. Air Force Institute of Technology, GGC/MA/73-122, 1973.

6. Smith, Rogers, E.: Effects of Control System Dynamics on Fighter Approach and Landing Longitudinal Flying Qualities, Vol. 1. Air Force Flight Dynamics Laboratory, AFFDL-TR-73-122, 1978.

7. Miller, D.P.; and Vinje, E.W.: Fixed-Base Flight Simulator Studies of VTOL Aircraft Handling Qualities in Hovering and Low-Speed Flight. Air Force Flight Dynamics Laboratory, AFFDL-TR-67-152, 1968.

8. Neal, Peter T.; and Smith, Rogers E.: An In-Flight Investigation to Develop Control System Design Criteria for Fighter Airplanes. Air Force Flight Dynamics Laboratory, AFFDL-TR-70-74, 1970.

9. Stengel, R.F.; and Miller, G.E.: Flight Tests of a Microprocessor Control System. Journal of Guidance and Control, Vol. 3, No. 6, Nov.-Dec. 1980, pp. 494-500.

10. Sammonds, R.I.; and Bunnell, J.W., Jr.: Flying Qualities Criteria for Wings-Level-Turn Maneuvering During an Air-to-Ground Weapon Delivery Task. AIAA Paper No. 80-1628. 1980 AIAA Atmospheric Flight Mechanics Conference, Aug. 11-13, 1980.

11. Hess, R.A.: Prediction of Pilot Opinion Ratings Using an Optimal Pilot Model. Human Factors, Vol. 19, No. 5, Oct. 1977, pp. 459-475.

Table I. Aircraft Configurations Analyzed

No.	Configuration				Reference
1	F-8 "Pitch Direct"	0.13 sec delay			2
2		.23			2
3		.33			2
4	"ISAS"	0.13 sec delay			2
5		.23			2
6		.33			2
7	"2D"				5,8
8	"5A"				5,8
9	"8A"				5,8
10	"9"				5,8
11	"10"				5,8
12	"11"				5,8
13	"12"				5,8
14	"3-1"				6
15	"4-1"				6
16	"6-1"				6
17	"PH-28"				7
18	"PH-29"				7
19	"PH-32"				7
20	"PH-35"				7
21	Princeton VRA	0.055 sec delay			9
22		.135			9
23		.255			9
24		.355			9
25		.455			9
	FSAA Wings-Level Turn (lateral-directional)	ζ_n	ω_n (rad/sec)	τ_D (sec)	10
26		1.4	15.0	0	10
27		1.4	2.0	0	10
28		2.0	8.0	0	10
29		0.7	6.0	0	10
30		0.5	4.5	0	10
31		0.3	4.5	0	10
32		1.4	4.5	0.49	10

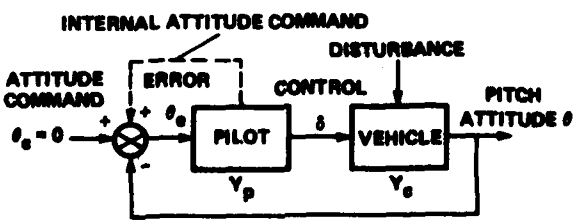


Figure 1. A pitch attitude regulation task.

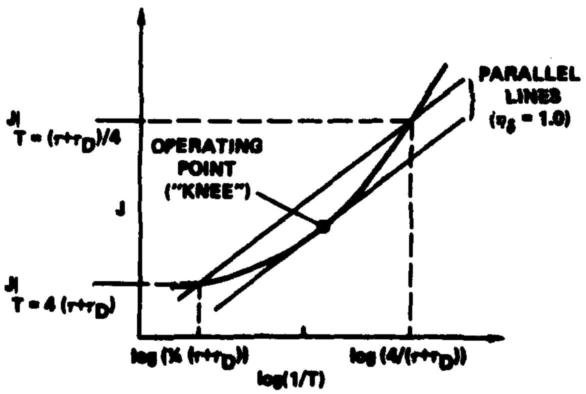


Figure 2. Selecting an "effective time constant" T.

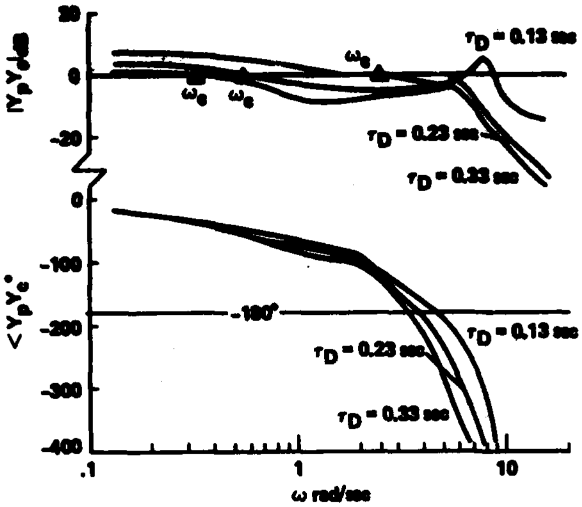


Figure 3. Pilot/vehicle dynamics for three configurations from Ref. 2.

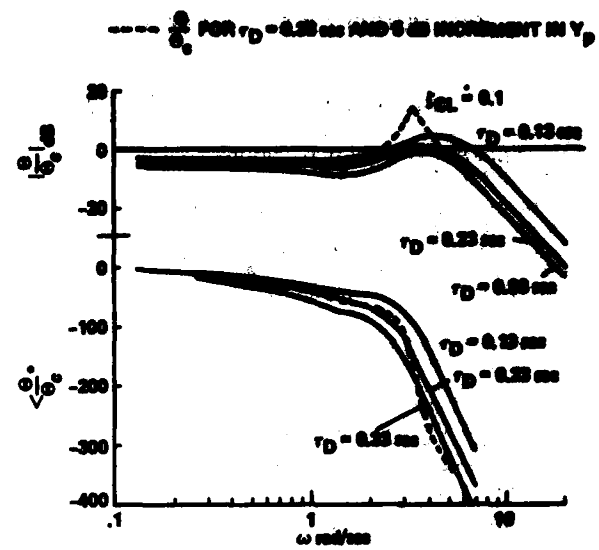


Figure 4. Closed-loop characteristics for three configurations from Ref. 2.

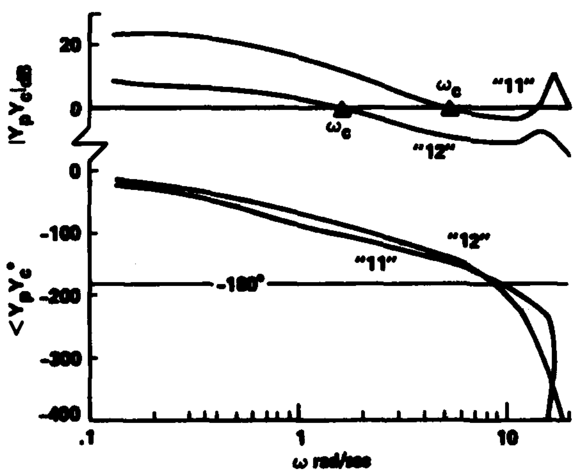


Figure 5. Pilot/vehicle dynamics for two configurations from Ref. 5.

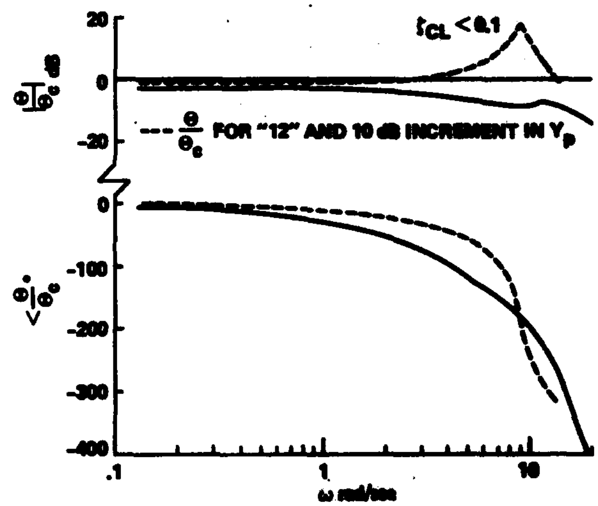


Figure 6. Closed-loop characteristics for a configuration from Ref. 5.

PIO TENDENCY RATING SCALE

DESCRIPTION	NUMERICAL RATING
NO TENDENCY FOR PILOT TO INDUCE UNDESIRABLE MOTIONS.	1
UNDESIRABLE MOTIONS TEND TO OCCUR WHEN PILOT INITIATES ABRUPT MANEUVERS OR ATTEMPTS TIGHT CONTROL. THESE MOTIONS CAN BE PREVENTED OR ELIMINATED BY PILOT TECHNIQUE.	2
UNDESIRABLE MOTIONS EASILY INDUCED WHEN PILOT INITIATES ABRUPT MANEUVERS OR ATTEMPTS TIGHT CONTROL. THESE MOTIONS CAN BE PREVENTED OR ELIMINATED BUT ONLY AT SACRIFICE TO TASK PERFORMANCE OR THROUGH CONSIDERABLE PILOT ATTENTION AND EFFORT.	3
OSCILLATIONS TEND TO DEVELOP WHEN PILOT INITIATES ABRUPT MANEUVERS OR ATTEMPTS TIGHT CONTROL. PILOT MUST REDUCE GAIN OR ABANDON TASK TO RECOVER.	4
DIVERGENT OSCILLATIONS TEND TO DEVELOP WHEN PILOT INITIATES ABRUPT MANEUVERS OR ATTEMPTS TIGHT CONTROL. PILOT MUST OPEN LOOP BY RELEASING OR FREEZING THE STICK.	5
DISTURBANCE OR NORMAL PILOT CONTROL MAY CAUSE DIVERGENT OSCILLATION. PILOT MUST OPEN CONTROL LOOP BY RELEASING OR FREEZING THE STICK.	6

Figure 7. The pilot-induced-oscillation rating scale.

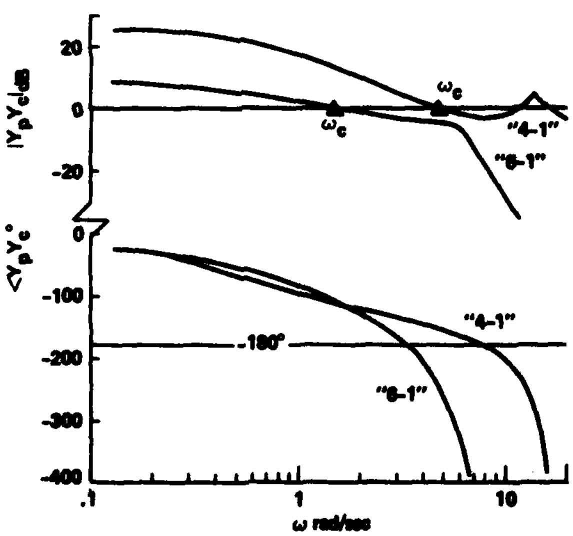


Figure 8. Pilot/vehicle dynamics for two configurations from Ref. 6.

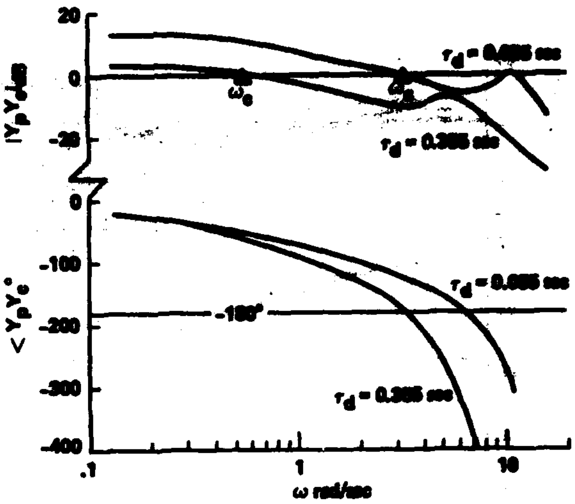


Figure 9. Pilot/vehicle dynamics for two configurations from Ref. 9.

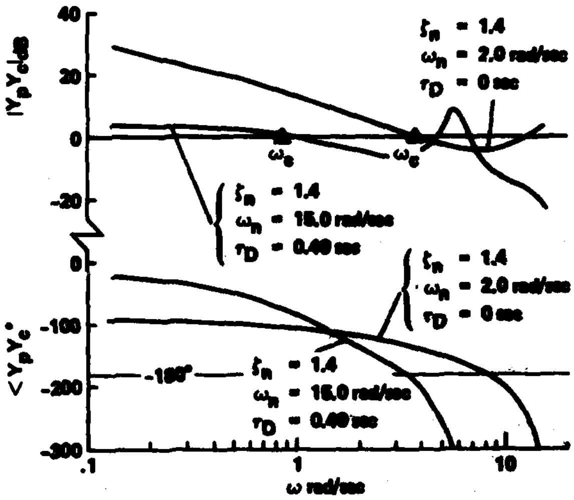


Figure 10. Pilot/vehicle dynamics for two configurations from Ref. 10.

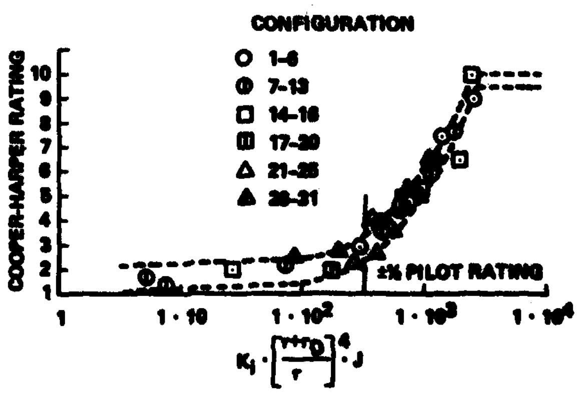


Figure 11. Cooper-Harper pilot ratings vs a proposed model-based metric.

SIMULATION FOR PREDICTING FLYING QUALITIES

Philip A. Reynolds
TIFS Program Manager
Flight Research Department
Arvin/Calspan Advanced Technology Center
P. O. Box 400
Buffalo, New York 14225

SUMMARY

This paper addresses the role of in-flight simulation in predicting and reproducing flying qualities. Reliable flying qualities predictions can be difficult. Measuring workload is technically challenging. There are psychological and political pressures on the evaluation pilot. The variety of dynamics and the number of parameters involved in airplane flying qualities are large. The tasks are frequently difficult to accurately simulate on the ground. Examples from several airplane development programs are presented to illustrate some of the faulty predictions that have been made with ground simulators. These examples provide flying qualities situations that can be used for more formal comparisons of ground simulation with in-flight simulation and for developing general hypotheses which could then be tested. Modern in-flight simulation capability is discussed emphasizing model-following accuracy, model complexity, and special effects such as artificial crosswinds, turbulence, and ground effects. Cost effectiveness is addressed. A role complementary to ground simulation is seen for in-flight simulation. The paper suggests that simulation planning could be improved by performing these formal experiments to define the predictive value of various ground simulator features such as large motion and high-quality visual cues.

LIST OF SYMBOLS

$e_{F_{as}}$	volts due to roll stick force
F_{as}	Roll stick force
F_{es}	pitch stick force
n_z	normal acceleration at the center of gravity
n_{z_p}	normal acceleration at the pilot's location
p	roll rate
p_c	roll rate command
$\dot{p}_{SA}$	roll acceleration in stability axes
q	pitch rate
$\bar{q}$	dynamic pressure
$\dot{r}_{SA}$	yaw acceleration in stability axes
s	Laplace operator
ΔT	thrust change from trim value
V	airspeed
α	angle of attack
δ_e	elevator deflection
δ_{e_c}	elevator deflection command
δ_{es}	pitch stick deflection
δ_{TH}	throttle deflection
θ	pitch angle
$[a]$	$\frac{s+a}{a}$, $a \neq 0$
$[b,a]$	$\frac{s+2bas+a^2}{a^2}$, $a \neq 0$
ms	milliseconds
FCS	flight control system
PIO	pilot-induced oscillation

INTRODUCTION

In this time of inflationary pressures and budget squeezes, it is more important than ever to do a better job of predicting flying qualities. In the later stage of an airplane's development, the high rate of expenditure makes even short delays very expensive. Hopefully, the day is past when extensive aerodynamic fixes are required. These would be disastrous to the modern development program. Nevertheless, if flying qualities problems are first uncovered in flight test, the cost can be substantial even if only flight control software changes are involved. Depending on the design, more costly firmware or hardware changes may be required.

For presently obscure reasons, the motivation to improve our knowledge of how to predict or reproduce flying qualities does not emanate strongly from the ground training field. Increased crew training in simulators is vital to both military and civil aviation. Good flying qualities reproduction which in the past has involved much of the same simulator technology as used in airplane development, should pay off in less flight training costs in the actual airplane. However, a relatively small portion of the pilot's typical training curriculum involves flying qualities in the short term, closed-loop sense. The presence of moving bases on training simulators attests to someone's belief that limited-motion acceleration-onset cuing is cost effective. Depending on the mission and the training objectives, this may or may not be true. On one hand, the discrete alerting side acceleration cue following an engine failure on takeoff may be quite important to training for emergencies. On the other, it is doubtful that much is to be gained in teaching fixed-wing instrument flying procedures by adding motion to a well-designed fixed-base simulator. Thus, the training cost effectiveness of motion cuing in ground simulators is controversial. High quality visual cuing, on the other hand, is vital for many training tasks.

The thesis of this paper is that predicting flying qualities involves risk and that the modern in-flight simulator can be a big help in reducing the risk. Much of this uncertainty arises from the fact that experiments have not been done which can provide the data to cost effectively partition an airplane development effort among simulator types and capabilities. A collection of experiences in several development programs is presented which illustrate the kinds of problems that prediction can encounter. These examples make the point that total reliance on ground simulation can be a mistake. The state-of-the-art in in-flight simulation is discussed to emphasize that this technology has matured enough to be routinely used in airplane development and even in part-task training. Future advances in this state-of-the-art are forecasted to greatly add to our understanding of flying qualities.

PREDICTING FLYING QUALITIES

The standard pilot comment after the first flight is that "it flew just like the simulator." If we could always be sure that this would be the result, we would have good reason to congratulate ourselves and flight pay would be a lot lower. Unfortunately, that statement is usually the beginning and ending of effort to correlate predictions with reality. Also the pilot may not be specifically referring to flying qualities by themselves but to a more overall impression of the airplane's behavior which may include rate of climb, stall speeds and other items of performance. Predicting flying qualities involves uncertainty. Some of this uncertainty lies in the process of flying qualities measurement. Some is due to psychological and political factors. Some is due to basic technical difficulty.

As is well known, flying qualities measurement involves closed-loop control performance and workload. Since flying qualities parameters can vary greatly at times without much effect on performance, we are obliged to measure workload. The workload, being largely mental, is not easy to measure. The task often has as much or more of an effect on the workload as the airframe/FCS dynamics. If the engineer has not defined the task in sufficient detail or clearly communicated that definition to the pilot, the flying qualities measurement may be drastically affected.

The major psychological and political factors are also well known to those who have been directly involved. The ideal evaluation pilot is an independent observer. He has achieved a measure of objectivity from experience in many different airplanes and can successfully place his own ego in the background. He is a trained evaluator who is able to recognize and be critical of flying qualities deficiencies. Unfortunately, the evaluation pilot can sometimes be far from this ideal. The political environment is rarely benign. The evaluation pilot frequently represents an organization and carries interests from that relationship to the simulator. He may be under considerable pressure if he has control difficulties, thus a high workload may go unreported. Or, he may be competing with other pilots to be the chosen one to fly the real vehicle.

The basic technical difficulty, aside from man-in-the-loop, is the number of dynamic parameters. Just when we think the vehicle is fully described for flying qualities analysis purposes, a new design exposes new parameters. Turbulence adds yet another dimension to the problem and there is no general agreement on what description of turbulence is to be used for flying qualities assessments. Theoretical models of the pilot have been developed to deal with the thicket of parameters and data that the empirical approach has encountered but their use is not widespread.

REPRODUCING THE TASK

Despite the difficulties, the aerospace industry has been moderately successful at predicting flying qualities using the Cooper-Harper rating scale to measure workload and a collection of simulation techniques to reproduce the task and cue environment. Further progress in this regard seems to be hampered by the lack of definitive data to select the proper simulator for each task. Practical knowledge on the roles of visual and motion cues is not plentiful and what there is is hard to generalize. When the decision on choice of simulator is made, the deciding factors appear to be availability, cost, and then task reproduction in about that order. Since the in-house simulator usually exists to satisfy the organization's needs, decision to use other facilities is rare. There is a tendency to accept a fixed-base simulation when the task is perhaps not well reproduced simply because it is readily available and cheap. A moving-base simulator may be used when excessive lags, insufficient degrees of freedom or insufficient travel cause it to be no better a predictor than the fixed-base simulator. The option to use an in-flight simulator may not even be considered for a variety of reasons. The technology is unfamiliar. The user is not aware of cost trades. The user expects task reproduction quality to be poor. The simulator is not as assessible or manageable. There is the understandable tendency to think well of one's own simulation facilities; "of course" they can reproduce the task well enough.

Can we learn from experience? Certainly if the last airplane developed using the company simulator flew as predicted and the new one is similar with only a few minor differences, the chances of poor prediction are minimized. Yet there are surprises in flight test - Why? Maybe we did not take the time to make predictions and check them so the simulation becomes "calibrated." Maybe the fact that the airplane was Level 1 in the simulator and Level 2 in flight was not attributed to poor simulator prediction. To perform a useful experiment in this regard, the simulator mathematical models must be substantiated and the evaluation pilot/pilots, after being carefully briefed on how to accomplish the task, must be reasonably objective. The fact remains that there are few documents we can consult that will tell us whether a given simulation of the task and airplane of interest will produce a reliable, unreliable or borderline representation of the flying qualities.

SOME EXAMPLES

The following are samples of research and development simulation work where the in-flight simulation results or the real vehicle differed markedly from the ground simulation. The general task in these examples is landing. Restricting the task to landing is not intentional and is probably not significant except that the landing task if pursued aggressively in a challenging environment is one of the most demanding of high-gain, closed-loop tasks.

1. Fighter Roll PIO on Takeoff and Landing, Refs. 1, 2, 3

Airplane developers have been unpleasantly surprised in recent years to find that roll control gradients determined by ground simulation can be too sensitive by large factors. An example of this is the mid-1970's development of a side-stick controlled fighter. After optimization on a fixed-base ground simulator, the system described in Appendix 1 was flown in the NT-33A in-flight simulator. The T-33 simulation, linearized except for breakout force, showed a tendency toward low-amplitude PIO on landing approach (takeoffs were not simulated). The roll sensitivity was reduced for the first flight as shown in Figure 1.

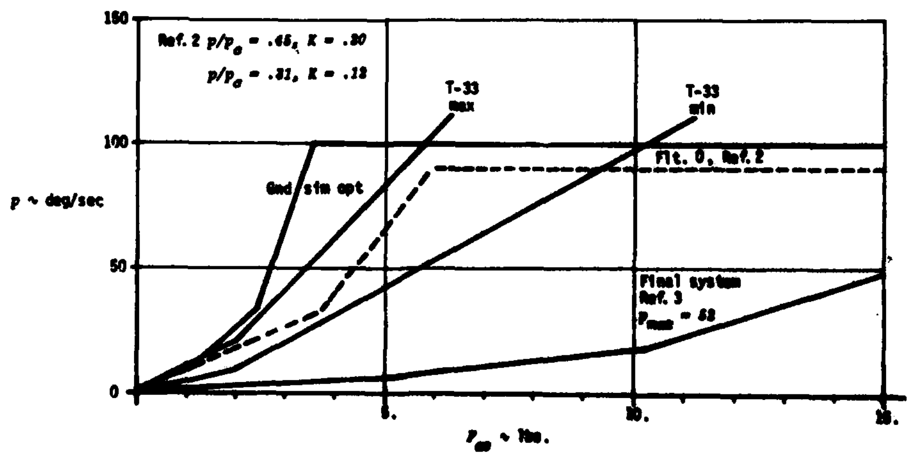


Figure 1. ROLL RATE EVOLUTION, V = 145 KIAS. (1 lb. = 4.45N)

During high-speed taxi tests prior to first flight, the prototype inadvertently became airborne. The left wing dropped rather rapidly and the pilot became immediately involved in a roll PIO which went through 10 cycles in 14.3 seconds (Ref. 2). The airplane was nearly lost. The pilot was able to open the roll control loop by continuing the unplanned takeoff to altitude to avoid wing strike. Analysis of the data showed full roll control and highly nonlinear behavior which undoubtedly aggravated the PIO as observed in the T-33. The stick shaper function, Figure 2, was reduced and the forward loop gain was also reduced. By the end of the flight test program, the roll rate per pound of stick force in the low roll rate control region had been reduced from the original ground simulator optimum value of about 13.0 deg/sec/lb (2.9 deg/sec/N) to about 1.2 (.27). The stick force to produce 30 deg/sec, a typical maximum value for landing, had been increased from 2 lbs (8.9N) to about 12 lbs (53.3N) neglecting breakout. The reduction in forward loop gain not only reduced the steady roll rate and increased the "roll mode" response time, but also (as pointed out by C. Chalk of Calspan Corporation in an unpublished analysis) proportionately decreased the initial roll acceleration following a sharp input. Therefore, although the steady roll rate per pound has been reduced by a factor of eleven, the initial roll acceleration per pound has been reduced by a factor of 18!

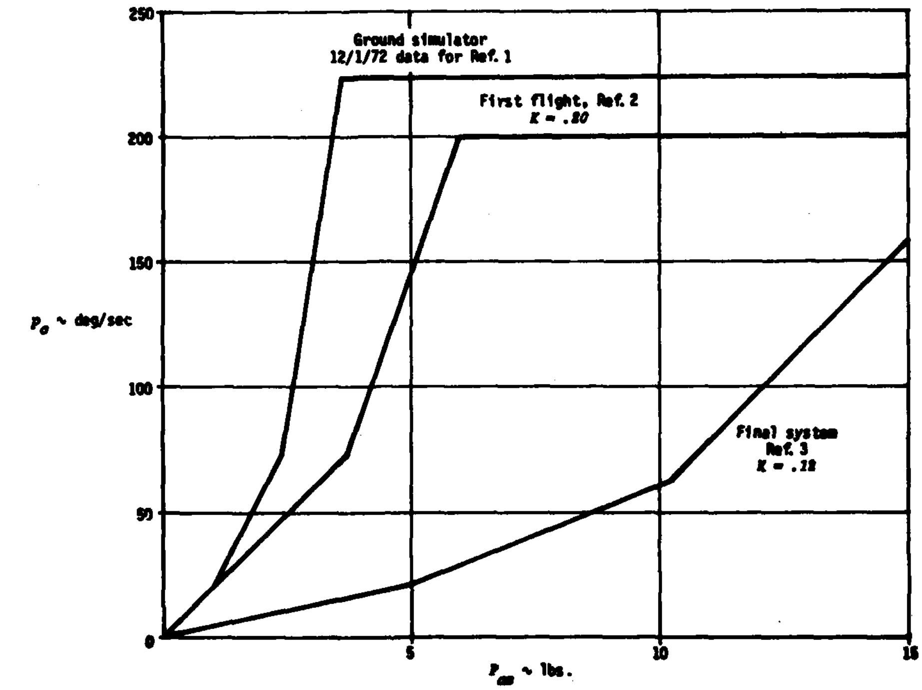


Figure 2. STICK COMMAND FUNCTION EVOLUTION (1 lb. = 4.45N)

Reference 4, pages 331-344, uses this example to come to the strong recommendation - "do not rely on fixed-base or limited motion simulators for compliance with this requirement" (roll axis control sensitivity).

2. Fighter Pitch PIO in Landing (Ref. 1)

In this new fighter development program, the NT-33A in-flight simulator was used by the Air Force to provide added confidence and pilot familiarity prior to first flight. The flight control system had been designed and developed with the benefit of a ground simulator with large motion capability. The results of the ground simulator investigation have not been documented in the literature but it can be assumed that the landing flying qualities of the fully operational FCS (Appendix 1) were expected to be Level 1. T-33 in-flight simulation, using the response feedback technique and approximations to the FCS overall pitch rate transfer function, revealed a landing PIO that was a serious problem to all the evaluation pilots. The pilot comments were typical of Level 3. The PIO susceptibility was not slight or the result of an off-nominal approach or unusual gusts. It was an obvious deficiency. In subsequent ground simulations on the same facility, it was reported that PIO was evident. It was also stated, after the fact, that the visual display was not of the quality judged sufficient for the landing task. Whether the landing techniques in the ground simulator and the in-flight simulator were the same is not a matter of record.

The FCS was modified and evaluated in the NT-33A as Level 1. The actual airplane as modified proved to be Level 1. Without in-flight simulation, the airplane might have been lost or severely damaged on the first flight. At best, a costly delay in the flight test program to work out the remedy would have resulted.

3. Supersonic Transport Relaxed Pitch Stability, Ref. 5, 6

This comparison between ground simulator and in-flight simulator experiments comes from a program to generate data on minimum acceptable longitudinal stability. The experiments consisted of twenty configurations made up of variations in C_{m_α}, $C_{m_q} + C_{m_{\dot{\alpha}}}$, and $\frac{d\gamma}{dV}$ (pitch static stability, pitch damping, and speed stability) on a baseline supersonic delta wing airplane with no stability augmentation. Although the original intent was not to compare ground with flight, these experiments constitute one of the few documented instances where the same dynamics were examined by the same pilots doing the same task. The ground facility was the FSAA at NASA/ARC - an engineering research simulator with a conventional 60 degree diagonal visual scene and a 6 DOF motion base with large lateral travel but normal travel otherwise. The in-flight simulator was the AFWAL/Calspan Total In-Flight Simulator (TIFS) with model-following, 6 DOF capability. The baseline airplane model was simplified for the in-flight simulator but was closely comparable.

Perhaps the most noticeable factor in comparing the two experiments is that the flying qualities prediction problem is very challenging for both methods of simulation. For example, consider the data scatter for Configurations 2, 3, 4 and 5 constituting the basic variation of C_{m_α} toward increasing instability (Table 1). The time to double amplitude of the dominant mode decreased from about sixty seconds in Configuration 2 to two seconds in Configuration 5. Yet along with this controlled variation, there were uncontrolled factors such as pilot learning and control technique and partially controlled factors such as turbulence magnitude and spectrum which disturbed the orderly progression of pilot ratings toward unflyable. The data analysts attempted to correct for these undesired effects in determining where the 6.5 boundary should be drawn. Effort to extract significant differences between ground and flight was outside the scope of the investigation, but both the ground and flight data analysts drew the 6.5 boundary at about the same level of time to double amplitude.

This example is interesting, among other reasons, because when most investigators would have and, in fact, did anticipate differences between ground and flight, there were no straightforward discrepancies. Raw averages of the ratings in Table 1 illustrate this point. Since the ground-based heave motion cue was only a small fraction of the in-flight cue and the pitch cue was missing entirely (through a simulator malfunction) one might conclude that motion cues play a quite minor role in this high-stress task. Perhaps this conclusion is erroneous but a more carefully controlled experiment is needed to demonstrate this.

TABLE 1

C-H RATING COMPARISON FOR EXAMPLE 3

	I		II		III		IV		Total Variation		Average Rating	
Conf.	Flt.	Grd.	Flt.	Grd.	Flt.	Grd.	Flt.	Grd.	Flt.	Grd.	Flt.	Grd.
2	4	---	8 to 10	6	---	5	6 to 7	6	4 to 10	5 to 6	6.5	5.7
3	6 to 7		6	4 to 5	9 to 10	6	6	7	6 to 10	4 to 7	7.0	5.8
4	5	---	7	5 to 7	---	6	7 to 8	9	5 to 8	6 to 9	6.5	7.0
5	10	---	---	9	10	10	7 to 10	10	7 to 10	9 to 10	9.5	9.7

4. Advanced Supersonic Transport Lateral Acceleration During Roll, Ref. 7, 8

This research is especially interesting from a flying qualities prediction viewpoint because a new parameter for transports was discovered - lateral acceleration at the cockpit during roll. The acceleration was large for this design because of the long fuselage, the angle of attack on landing approach, and the lateral-directional control system design. The design included high roll damping and used yaw control to maintain sideslip angle small during roll. Because the airplane was designed to roll about the flight path, which was some 36 feet (11.m) below the pilot at the approach angle of attack, the linear side acceleration due to roll angular acceleration was high. This was further aggravated by the high initial roll acceleration resulting from the small roll mode time constant. The data in Appendix 1 shows that for a coordinated turn entry, where $r_{s_A} = n_y = 0$ just after roll control input, there was .02 g's per deg/sec^2 roll acceleration at the pilot's station. For a 1 lb. step input at the control wheel grip, the steady state roll rate is 1.7 deg/sec and the peak roll acceleration is about 4.7 deg/sec^2 giving an abrupt lateral acceleration of .09 g's - a very distracting jolt.

When the airplane was evaluated on a fixed-base ground simulator, this side acceleration was completely unknown to the pilot. Not only was the motion cue missing but also the ball indicator, driven by the much smaller side acceleration at the center of gravity, gave no indication of the cockpit motion. The overall pilot rating for landing was 2 in the ground simulator and 7 to 8 in the in-flight simulator. The motion environment in flight was reported from a ride quality viewpoint to be disturbing to the point where the pilot found it virtually impossible to fly the airplane smoothly. Manikin effects probably were a factor. (It is worth noting that the real SST might behave differently, either better or worse, from both of these simulations due to the presence of wing and fuselage structural flexibility. Structural dynamic motions were not included in the research described here).

5. Space Shuttle Orbiter Landing Pitch Control, Ref. 9, 10

The final landing of the shuttle Approach and Landing Test (ALT) program was made on the 15,000-foot main runway at Edwards AFB. Just prior to touchdown a PIO occurred in pitch followed by a bounce and then a PIO in roll before the vehicle finally settled on the runway. The flight control system had been designed using extensive fixed and moving base simulation. No PIO problem had been noted. Several investigations involving analysis and simulation were subsequently undertaken. The simulations involved the FSAA and the TIFS along with various other facilities. A description of the TIFS experiment is given in Reference 9 and the results are analyzed in Reference 10.

The pitch axis characteristics are given in Appendix 1 in approximate form. The hand controller on ALT had a light gradient with strong centering. The pitch rate feedback FCS employed positive elevon feedback to produce a proportional-plus-integral transfer function in the forward loop so that stick deflection away from center commanded a pitch rate and return to center produced attitude-hold. The response was fairly slow with significant time delay. It is also important to note that the pilot station was near the center of percussion of elevon aerodynamic input so that the initial vertical acceleration following abrupt pitch input was very small.

PIO during the TIFS simulations was a subtle phenomenon although the PIO susceptibility was not. The pilot ratings ranged from 2 to 8 depending on task definition, pilot training, pilot technique, gust disturbances and necessity for corrections late in the flare. Indeed, the results on the first five actual ALT landings where the first four were smooth and uneventful suggest that the problem might be difficult to predict. (It is probably significant that these first four were lakebed landings with no requirement for rapid corrections near the ground). However, in-flight simulation provided flying qualities like the behavior of the ALT on its final flight. To quote from Reference 10, page 35:

> "TIFS results indicate the Orbiter's longitudinal handling qualities during the landing phase are worse than what was realized by the AFFTC prior to the TIFS program. The increased pilot gain provided by the actual flight environment of the TIFS simulator brought out longitudinal handling qualities deficiencies in the Orbiter that were less apparent on previous ground-based simulations. Numerous longitudinal oscillations of the type experienced during the landing phase of ALT Free Flight 5 were encountered during the TIFS program. Several larger amplitude and divergent pilot-induced oscillations were also encountered."

Subsequent to these investigations in 1978, additional ground simulations of the orbiter landing have been carried out on the NASA/Ames Vertical Motion Simulator (VMS) and in the TIFS. These tests have shown that there are also significant differences in PIO susceptibility between the VMS with its large vertical travel and high bandwidth and the TIFS for the precision landing flare task, with the in-flight simulator providing the more faithful prediction of the orbiter PIO problems.

6. Advanced Subsonic Transport Direct Lift Control, Ref. 11, 12

Over the past several years, the Dutch National Aerospace Laboratory (NLR) has been conducting a carefully controlled series of advanced transport evaluations. Flight control systems including rate-command, attitude-hold, stability augmentation and direct lift control were studied using a moving base ground simulator and an in-flight simulator. The latter was used as a means to verify and calibrate the ground results, particularly in the flare and touchdown part of the landing task. There were several similarities and several differences in the results. The familiar difference of touchdown sink rates was apparent with the ground simulator averaging 4 fps (1.2 mps) while the average in flight was 2 fps (.6 mps). As the basic pitch response parameters were varied, the pilot rating variations were comparable. However, the variations with direct lift control were optimistic in the ground simulator. As shown in Fig. 3, the

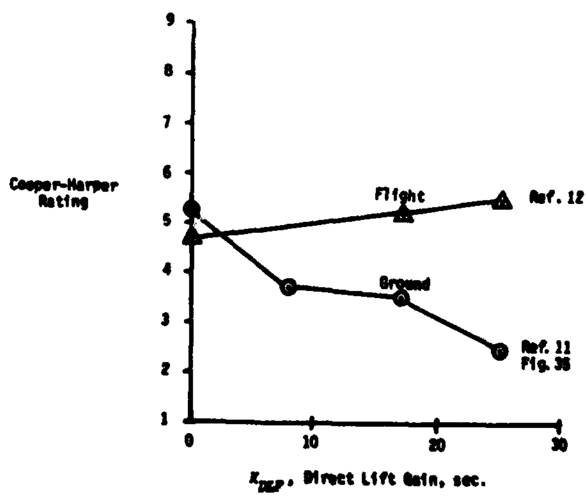


Figure 3. DIRECT LIFT EVALUATION (EXAMPLE 6)

rating continued to improve with increase in direct lift control gain in the ground simulator and showed a reverse trend in flight due to the objectionable vertical accelerations. In fact, the point plotted at $K_{DLF} = 25$ in flight was not formally evaluated because pre-evaluation showed the rating would be poor enough not to be of further interest. It is this configuration which is described in Appendix 1.

It should be noted that the aerodynamics described in Reference 11 describe a backside speed variation with flight path angle, and the authors improved the pilot ratings by half a point extrapolating to a frontside operation. The flight results were for frontside operation and thus are comparable with the extrapolated ground results shown in Fig. 3. It should also be noted that the pilots' speed control complaints were louder and more numerous in flight than on the ground even though one might predict the opposite.

GENERALIZATIONS

The above examples are not results of rigorous experiments comparing ground with flight but perhaps they point the way to fruitful areas of investigation and to general hypotheses which need testing. We might ask the following questions:

a) Is it true that, lacking the strong phase lead information of roll acceleration and acceleration rate, the ground simulator will almost always lead to overly sensitive roll control gain selection? Can peripheral visual cues help?

b) Is there a similar effect in the pitch axis?

c) Is the statement that if the flying qualities are good in the ground simulator they will be good in flight a valid generalization? The examples quoted above tend to refute this.

d) Is the statement that if the flying qualities are poor in the ground simulator (i.e., the pilot workload and compensation are high) the prediction will be unreliable a valid generalization? The experiment from which example 3 is taken shows that although the scatter was great, the stability boundary was correctly predicted.

e) Can we explain, in general terms, when complete motion cuing improves flying qualities, when it leads to combined flying/ride qualities difficulties, when it degrades flying qualities?

f) Is it useful to distinguish between the simulator transfer function from computed motion to platform motion and the pilot transfer function from platform motion to control action? Can the presence of PIO's in flight and not on the ground be due more to "cue gain" differences than "pilot gain" differences?

g) Is there a cuing threshold (possibly a function of vehicle and task) below which the pilot will tend not to close that particular loop even though the cue may be perceptible? If so, this may lead to a rationale for deciding when adding ground-based limited motion to a simulation plan is not cost effective.

h) Can flying qualities degradation in turbulence be correctly predicted on the ground or are the motion cues simply inadequate?

MODERN IN-FLIGHT SIMULATION CAPABILITY

How close can the modern in-flight simulator come to predicting the flying qualities of a given airplane? The flying qualities are usually accurately predicted if the math model is accurate. Model following lags are minimized by feedforward techniques and occasionally compensated by model lag reduction. Aerodynamic control surface bandwidths are in the neighborhood of 3.to 10.Hz.. Elevator and rudder inputs act through the stiff fuselage structure to produce pitch and yaw accelerations at the pilot's station. The ailerons typically act through wing bending modes of lower frequency to produce roll accelerations. The roll system will be somewhat slower if a single actuator, mounted at the centerline, acts through cables. However, even for an in-flight simulator the size of the TIFS, the c.g. roll acceleration starts only 80 ms after the start of aileron deflection and the actuator and cable lags are small giving an overall roll acceleration lag of about 100 ms. Other motion acceleration lags are about the same or less except for the fore-and-aft force control through the throttle controls which includes the lag of the propjet engine thrust to throttle motion input. The velocity model-following performance is quite good because the motion bandwidth requirement is low. Feedforward from model-commanded accelerations improves the model-following performance of the feedback error comparison system to eliminate all but actuation system and structural flexibility lags. When the model contains serial lags such as a smoothing filter in the command path, they can be modified to account for even these simulation lags.

Because in-flight simulation, with its real world cues, serves, in many instances, the role of the final authority, the cuing accuracy receives much attention. Frequency response information for each commanded degree of freedom are evaluated for amplitude flatness and phase lag minimization. The motion response coupling (heave response to pitch command, for example) is minimized. Motion time histories of the model and the simulator are overlayed to examine initial lag, rise time, overshoot and steady state errors.

Model complexity has increased ste... in recent years. With the advent of small, powerful airborne general purpose digital computers, it is no longer necessary to linearize or to omit small terms from the equations of motion. Configuration change is faster and easier to accomplish, flight control system logical features can be reproduced, mode switching and failure mode transients can be duplicated and accuracy is improved.

The motion limits dictated by the in-flight simulator basic performance, maneuverability and control power are compared with the model characteristics to determine simulation limits. For example, the simulator direct vertical force control moves roughly in proportion to the ratio of the simulator and model n_z/α's. The permissible control deflection at a given airspeed is a function of vertical load factor giving rise to an operational area on a plot of deflection limit versus load factor. The intersections of the operating line defined by $(n_z/\alpha)/(n_z/\alpha)_m$ and the boundaries of permissible area determine the positive and negative load factor limits that can be used in simulation.

Simulation under conditions where the simulator and model airspeeds are not the same has been worked out for several different motion matching criteria. The criterion perhaps of most interest is matching angular rates and linear velocity perturbations. The up-and-away flying qualities are reproduced for small departures from unaccelerated flight. The smaller the airspeed difference, the larger can be the perturbations. This technique is used for tasks such as in-flight refueling.

Several special features make landing simulation more accurate. By trimming in a wings-level sideslip with zero side acceleration using the side force capability, the simulator can introduce crosswind to the landing task. Likewise, the effects of an unwanted real crosswind can be removed by using the same technique. Using a sensitive radar altimeter signal, the crosswind can be made a function of altitude. Artificial head and tail winds can be introduced through the simulator's ability to adjust trim attitude and flight path independent of airspeed. The existing real turbulence is measured and fed to the model. The model-following system tends to suppress the TIFS response and substitute the model response. Random turbulence is also produced by playing tape recorded signals of the desired statistical behavior into the model equations. The amplitude of various components can be changed as a function of altitude using the radar altimeter signal. This signal is also used to produce the forces and moments due to ground effect which are important in defining the landing task in the flare. Proper touchdown eye height is produced by computing when the model wheels contact the ground and cuing the pilot. Actual touchdowns are used when the jolting penalty for a poor landing is needed to motivate the pilot or to expose the effects of high gain behavior. Discrete gusts are sometimes programmed into the landing task to force the pilot into aggressive control action which may trigger PIO.

The cost per evaluation hour in an in-flight simulator is comparable with that of the large motion ground simulators. Its productivity for the landing task in terms of approaches per hour is not as great because of the "reset" capability in the ground simulator. Perhaps, however, in-flight simulator technology for generating real and complete motion, combined with the rapidly improving capability to generate bright, high detail, wide angle artificial visual scenes could produce an even more powerful simulation device for predicting flying qualities with improved cost per run!

Frequently, the point is made that the in-flight simulator is capable of producing more accurate flying qualities prediction because it is real. This point is probably valid. The pilot finds it natural to fly with the same technique and concern that he would use in the actual airplane. It is true that the safety pilot is always there, so in very critical situations, the pilot's "gain" (urgency, intensity) may still not be as high. Also properly motivated test pilots probably fly the ground simulator with the same concern to perform well as they do their aircraft. Thus, it seems unlikely that pilot respect for the simulation and concern for the consequences of his control action would explain many of the differences found between ground and flight. Yet these differences continue to appear.

ROLE OF IN-FLIGHT SIMULATION

As illustrated by the examples in this paper, in-flight simulation has been used to complement analysis and ground simulation. The major use, aside from research, has been for final system verification, but it has also been a crew trainer prior to first flight and a test bed for flight hardware. Experience with new airplane development support suggests that earlier use would be more cost effective. If FCS design deficiencies had been discovered earlier, time and money would have been saved. In Example 6, it is obvious that early knowledge of the normal acceleration problem has potentially large economic impact. Although a supersonic transport is not currently under development, it is clear that early awareness of the lateral acceleration environment from Example 4 has added greatly to understanding the constraints on lateral control design and would save development dollars on any such program.

As Example 1 shows, there is also a role for in-flight simulation after the first flight. The roll control gain and maximum authority was changed considerably during the test program as shown in Figures 1 and 2 with all development done on the test airplane. Since the test airplane is much more expensive to fly than the simulator, and the cut-and-try mode of fixing FCS problems on the test airplane is a very costly route to the final solution, the combination of in-flight simulation and test airplane flying should be explored.

We do not fully understand why ground simulation did not correctly predict the flying qualities in some of these examples. If formal experiments show the ground-based cues to be insufficient for some tasks and sets of flying quality parameters, the in-flight simulator should be involved at all stages of airplane development where these cases are studied.

The USAF owns the TIFS and the T-33, and currently makes them available to any organization in the industry. These facilities are accessible and can operate any place in the country. Through telemetry link, the TIFS could operate as a motion base for a model already mechanized in the contractor's

ground simulator. The cost of preparing another simulation is thus minimized.

The examples of this paper are certainly not the only instances of poor prediction from ground simulators. Likewise, the picture painted here does not include airplanes for which the prediction was accurate and unequivocal. Simulation technology is evolving rapidly in modeling accuracy and visual cuing to deal with real mission complexity. We should also do the experiments which will sharpen our ability to choose the proper simulation facility for flying qualities evaluation. The formal experiments could start by imbedding the examples here in a related spectrum of airplane dynamics and FCS designs. As far as is possible, all factors affecting flying qualities should be identical from ground to flight except, of course, the motion and visual cues. Careful attention should be paid to task definition and evaluation pilot briefing. In-flight simulation should be used for the flight portion so that the model behavior can be matched to the ground simulator without costly flight testing and model development. The results of this kind of simulation research could herald a new era of improved airplane flying qualities.

REFERENCES

1. Hall, G. W., and Harper, R. P., Jr.: "In-Flight Simulation of the Lightweight Fighters;" AIAA Paper No. 75-985; AIAA Aircraft Systems Technology Meeting; August 1975.

2. Smith, J. W.: "Analysis of a Lateral Pilot-Induced Oscillation Experienced on the First Flight of the YF-16 Aircraft;" NASA TM 72867; September 1979.

3. Eggers, J. A., and Bryant, W. F., Jr.: "Flying Qualities Evaluation of the YF-16 Prototype Lightweight Fighter;" AFFTC-TR-75-15; July 1975.

4. Hoh, R. H., et. al.: "Proposed MIL Handbook - Handling Qualities of Piloted Airplanes;" AFWAL TR-82- ; December 1981; Preliminary version.

5. Snyder, C. T., et. al.: "Motion Simulator Study of Longitudinal Stability Requirements for Large Delta Wing Transport Airplanes During Approach and Landing with Stability Augmentation System Failed;" NASA TMX-62,200; December 1972.

6. Wasserman, R., et. al.: "In-Flight Simulation of Minimum Longitudinal Stability for Large Delta-Wing Transports in Landing Approach and Touchdown;" Vols. I and II; AFFDL-TR-72-143; February 1973.

7. Grantham, W. D., and Nguyen, L. T.: "Recent Ground-Based and In-Flight Simulator Studies of Low-Speed Handling Characteristics of Supersonic Cruise Transport Aircraft;" AIAA 1977 Atmospheric Flight Mechanics Conference; AIAA 77-1144; August 1977.

8. Weingarten, N. C.: "An Investigation of Low Speed Lateral Acceleration Characteristics of Supersonic Cruise Transports Utilizing the Total In-Flight Simulator (TIFS);" Calspan Report No. 6241-F-2; March 1979.

9. Weingarten, N. C.: "In-Flight Simulation of the Space Shuttle Orbiter During Landing Approach and Touchdown in the Total In-Flight Simulator (TIFS);" Calspan Report No. 6339-F-1; September 1978.

10. Kirsten, P. W.: "In-Flight Simulation of the Space Shuttle Orbiter Low Speed Handling Qualities Using the USAF/Calspan TIFS Airplane;" U.S. Air Force Flight Test Center, Office of Advanced Manned Vehicles; Office Memo, October 1978.

11. Mooij, H. A., deBoer, W. P., and van Gool, M.F.C.: "Determinations of Low-Speed Longitudinal Maneuvering Criteria for Transport Aircraft with Advanced Flight Control Systems;" Nationall Lucht-en Ruimtevaartlaboratorium; NLR TR 79127 L; December 1979.

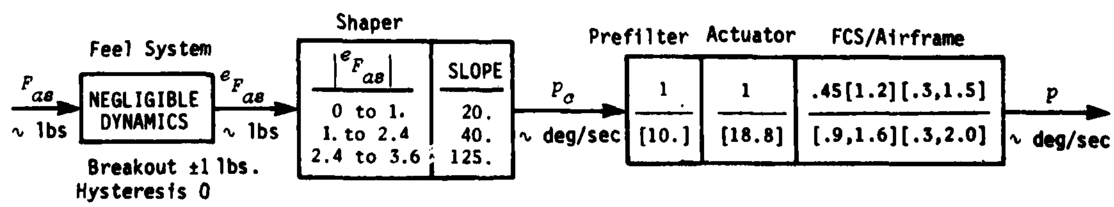
12. Weingarten, N. C.: "Flying Qualities Experiment Performed for the National Aerospace Laboratory (NLR), The Netherlands, in the Total In-Flight Simulator;" Calspan Report No. 6645-F-4; March 1981.

APPENDIX 1

DYNAMIC CHARACTERISTICS IN EXAMPLES
(1 lb. = 4.45N; 1 in. = 2.54 cm)

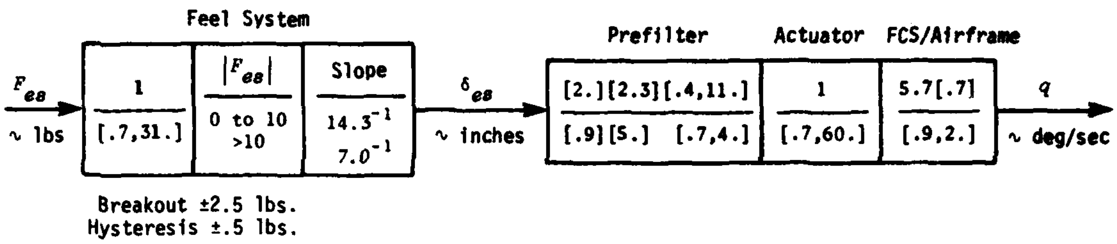
1. Fighter roll PIO on takeoff and landing, References 1 and 2

V = 145 kts.



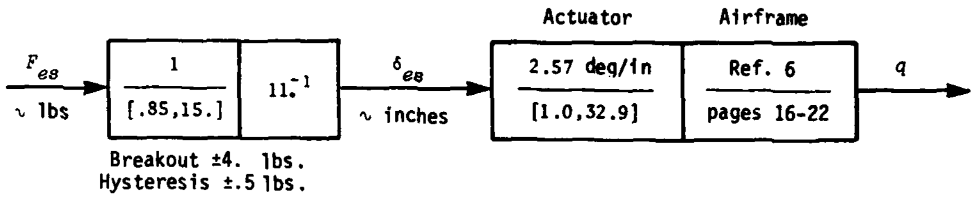
2. Fighter pitch PIO in landing, Reference 1

V = 118 kts.



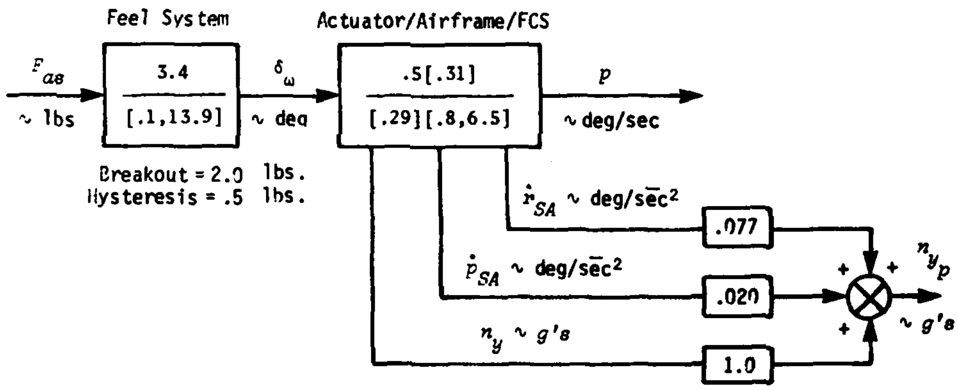
3. Supersonic transport relaxed pitch stability, Reference 6

V = 160 kts.



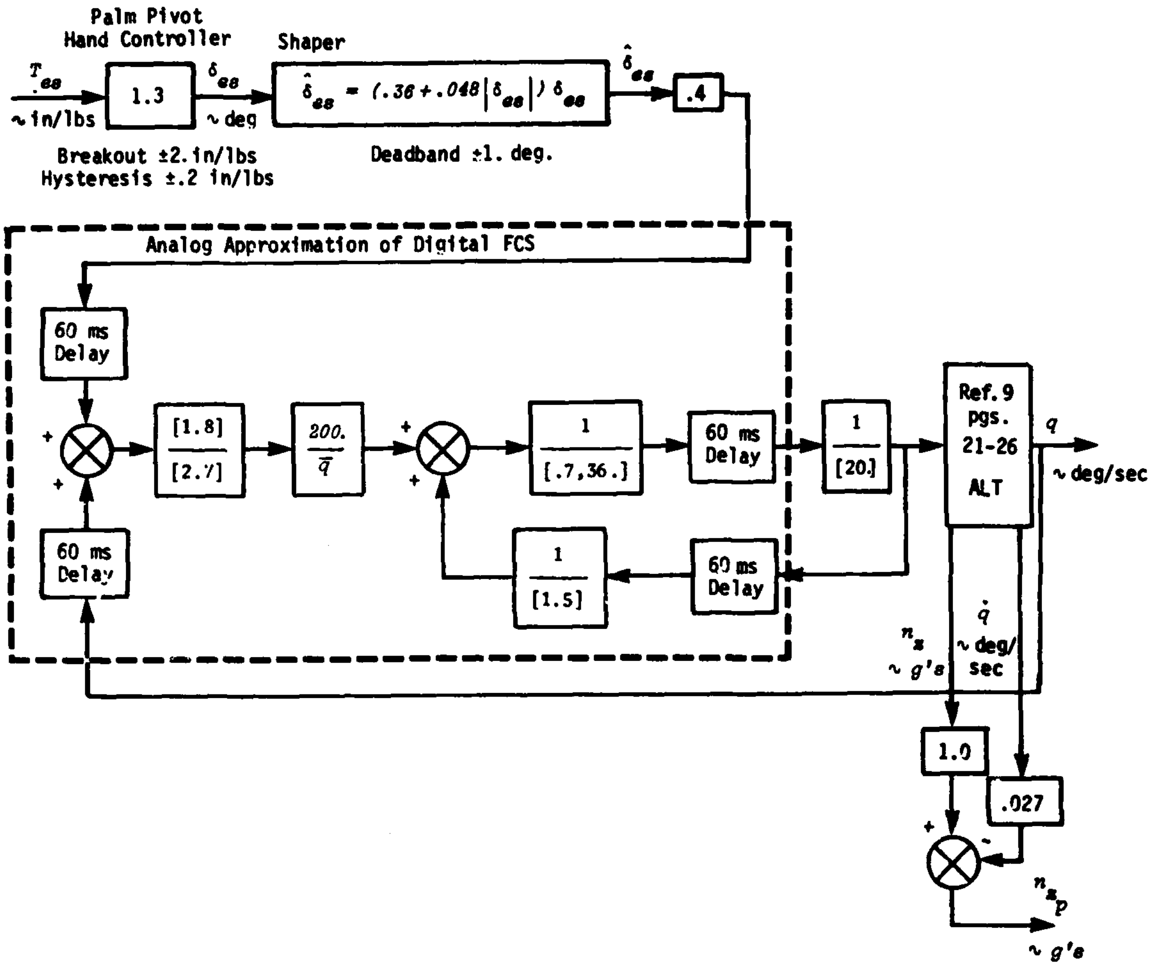
4. Advanced supersonic transport lateral acceleration during roll, References 7 and 8

V = 153 kts., α = 8.2 deg.



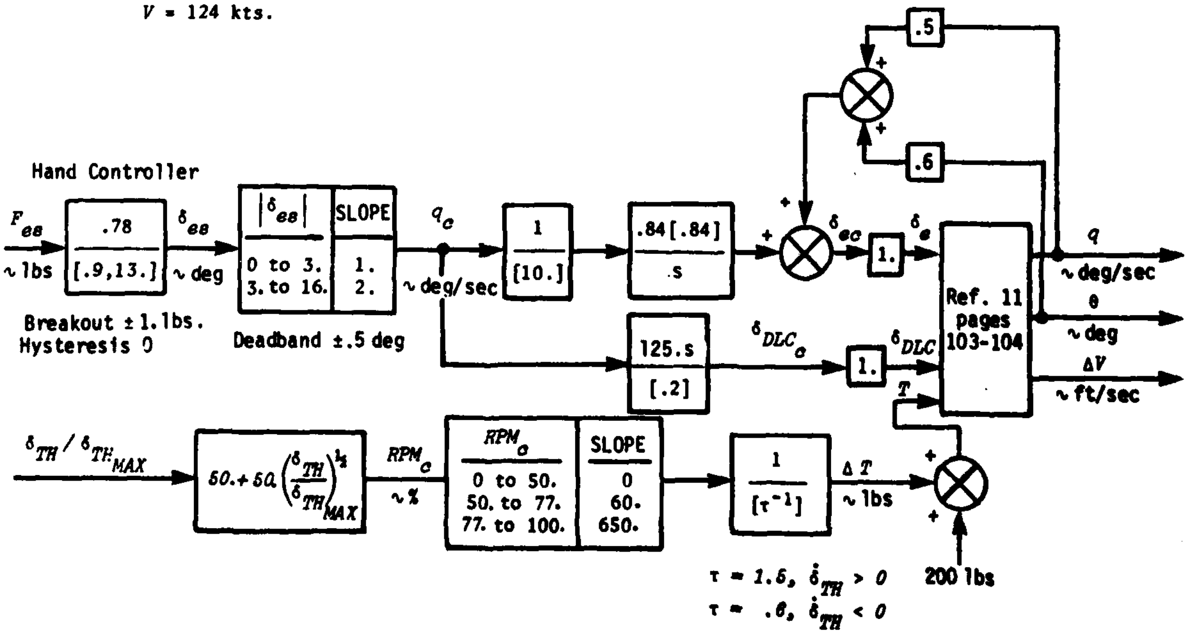
5. Space shuttle orbiter landing pitch control, Reference 9

V = 290 kts. to 180 kts. (at touchdown)



6. Advanced subsonic transport direct lift control, References 11, 12

V = 124 kts.



REPORT DOCUMENTATION PAGE

1. Recipient's Reference	2. Originator's Reference	3. Further Reference	4. Security Classification of Document
	AGARD-CP-333	ISBN 92-835-0313-8	UNCLASSIFIED

5. Originator Advisory Group for Aerospace Research and Development
North Atlantic Treaty Organization
7 rue Ancelle, 92200 Neuilly sur Seine, France

6. Title

CRITERIA FOR HANDLING QUALITIES OF MILITARY AIRCRAFT

7. Presented at

8. Author(s)/Editor(s)	9. Date
Various	June 1982

10. Author's/Editor's Address	11. Pages
Various	336

12. Distribution Statement This document is distributed in accordance with AGARD policies and regulations, which are outlined on the Outside Back Covers of all AGARD publications.

13. Keywords/Descriptors

Military aircraft
Flight characteristics
Maneuverability
Criteria

14. Abstract

These Proceedings consist of the papers presented at the AGARD Flight Mechanics Panel Symposium on Criteria for Handling Qualities of Military Aircraft held in Fort Worth, US, 19–22 April 1982. Topics covered include: present status of criteria, gains achieved in the Seventies and future prospects, criteria for flight at high angle of attack, special problems and techniques for handling qualities determination.

AGARD Conference Proceedings No.333
Advisory Group for Aerospace Research and Development, NATO
CRITERIA FOR HANDLING QUALITIES OF MILITARY AIRCRAFT
Published June 1982
336 pages

The Proceedings consist of the papers presented at the AGARD Flight Mechanics Panel Symposium on Criteria for Handling Qualities of Military Aircraft held in Fort Worth, US, 19–22 April 1982. Topics covered include: present status of criteria, gains achieved in the Seventies and future prospects, criteria for flight at high angle of attack, special problems and techniques for handling qualities determination.

ISBN 92-835-0313-8

AGARD-CP-333

Military aircraft
Flight characteristics
Maneuverability
Criteria

AGARD Conference Proceedings No.333
Advisory Group for Aerospace Research and Development, NATO
CRITERIA FOR HANDLING QUALITIES OF MILITARY AIRCRAFT
Published June 1982
336 pages

The Proceedings consist of the papers presented at the AGARD Flight Mechanics Panel Symposium on Criteria for Handling Qualities of Military Aircraft held in Fort Worth, US, 19–22 April 1982. Topics covered include: present status of criteria, gains achieved in the Seventies and future prospects, criteria for flight at high angle of attack, special problems and techniques for handling qualities determination.

ISBN 92-835-0313-8

AGARD-CP-333

Military aircraft
Flight characteristics
Maneuverability
Criteria

AGARD Conference Proceedings No.333
Advisory Group for Aerospace Research and Development, NATO
CRITERIA FOR HANDLING QUALITIES OF MILITARY AIRCRAFT
Published June 1982
336 pages

The Proceedings consist of the papers presented at the AGARD Flight Mechanics Panel Symposium on Criteria for Handling Qualities of Military Aircraft held in Fort Worth, US, 19–22 April 1982. Topics covered include: present status of criteria, gains achieved in the Seventies and future prospects, criteria for flight at high angle of attack, special problems and techniques for handling qualities determination.

ISBN 92-835-0313-8

AGARD-CP-333

Military aircraft
Flight characteristics
Maneuverability
Criteria

AGARD Conference Proceedings No.333
Advisory Group for Aerospace Research and Development, NATO
CRITERIA FOR HANDLING QUALITIES OF MILITARY AIRCRAFT
Published June 1982
336 pages

The Proceedings consist of the papers presented at the AGARD Flight Mechanics Panel Symposium on Criteria for Handling Qualities of Military Aircraft held in Fort Worth, US, 19–22 April 1982. Topics covered include: present status of criteria, gains achieved in the Seventies and future prospects, criteria for flight at high angle of attack, special problems and techniques for handling qualities determination.

ISBN 92-835-0313-8

AGARD-CP-333

Military aircraft
Flight characteristics
Maneuverability
Criteria